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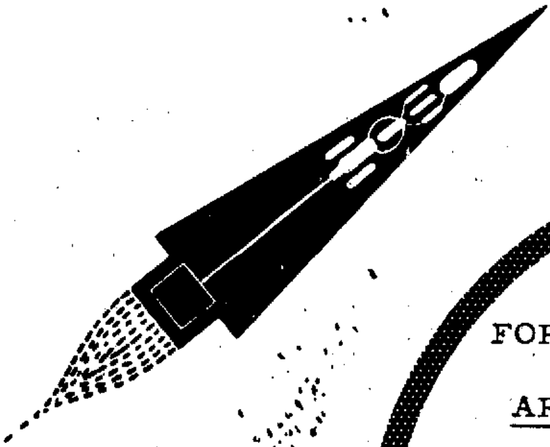

FINAL REPORT

FOR THE PERIOD 3/1/61 - 8/31/61

ARC JET APPLICATION STUDY

UNDER CONTRACT NAS 5-1034

CONTROL NUMBER GS 2264

FOR

THE NATIONAL AERONAUTICS AND

SPACE ADMINISTRATION

SEPTEMBER 25, 1961

FLIGHT PROPULSION LABORATORY DEPARTMENT

GENERAL ELECTRIC

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ABSTRACT

Studies of space mission applications for the Snap 8 - 30 KW Arc Jet propulsion system are described. The major mechanical and electrical system components required for operation of the systems as a complete propulsion system are described as are overall vehicle configurations utilizing the Atlas - Centaur booster. Specific candidate missions are identified and the performance capabilities of the propulsion system for each of the missions determined. The NASA and DOD space programs which would be suitable for the propulsion system are identified.

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1. INTRODUCTION

The objectives of this study program were to identify component requirements for operating the 30 KW Arc Jet Engine from the Snap 8 Power Supply, to determine propulsion system performance capabilities for selected candidate missions, to identify the most attractive space program applications for the propulsion system, and to recommend propellant selection. Analytical and experimental arc jet engine characteristics have been obtained from the parallel Arc Jet Engine Development Program - NASA Contract No. NAS 5-506 - and utilized in the study. Additional data on the Snap 8 Power Supply and the Centaur and Saturn Boost Vehicle specifications have been obtained from NASA Headquarters. Information on NASA program plans have been obtained from discussions with Goddard Space Flight Center and G. E. Missile and Space Vehicle Department personnel.

A summary of the results and conclusions obtained for each section is included in Section 2.0.

The Snap 8 and 30 KW Arc Jet Characteristics and specifications are described in Section 3.0. These data are utilized to define and to develop the major mechanical and electrical systems required for operation as a complete propulsion system.

In Section 4.0, vehicle design criteria are established and used to develop three arc jet vehicle configurations suitable for orbital propulsion after injection into a low altitude parking orbit by the Atlas - Centaur boost vehicle.

Theoretical engine performance characteristics for both hydrogen and ammonia propellants are described in Section 5.0. These data are compared with experimental test points obtained from the General Electric XT-761 Arc Jet Engine.

The selected candidate missions are introduced in Section 6.0 and the vehicle configurations required for each identified. Typical mission trajectory profiles are described and parametric mission performance capabilities indicated. Also included are comparative mission performance capabilities of hydrogen and ammonia for two of the candidate missions.

In Section 7.0, the most appropriate space program applications for the arc jet propulsion system are described and the specific engine performance capabilities indicated.

The Appendix - Section 9.0 - contains the trajectory equations utilized in generating the parametric mission performance capabilities and a description of their development.

Supplementary references are included in Section 8.0.

2. RESULTS AND CONCLUSIONS

The results of the Arc Jet Application Study - NASA Contract No. NAS 5-1034 - are summarized in the following sections. The major results achieved in the individual studies of system components, conceptual design, engine performance, mission analysis, and applications are indicated. The conclusions derived from these results are discussed and utilized to formulate specific recommendations.

2.1 Summary of Results

2.1.1 System Components

The major mechanical components required for operation of the propulsion system are the propellant storage, propellant feed, and thrust vectoring systems. Propellant storage system construction should utilize a double - walled design to provide for meteorite protection and thermal insulation in the atmosphere. Thermal insulation in space will be provided by the use of multi-layers of aluminum foil around the outside of the tank.

Propellant feed will be accomplished by controlled vapor boil-off in the propellant storage tank. Propellant feed lines will also serve as the power transmission lines from the Snap 8 to the engine. A double gimbal arrangement will be used to provide thrust vectoring in two dimensions for achieving yaw and pitch control of the vehicle. Roll control will be accomplished by the use of four 1 KW arc jet engines operated from the main power supply.

The major electrical components will include a transformer, inductances, switchgear, and a parasitic load. The transformer will utilize a three phase wye-to-delta connected auto-transformer design with hollow conductors for propellant flow to achieve cooling. The required inductances will be obtained from the leakage reactance of the transformer.

Vacuum switch-gear will be utilized for power management in order to avoid any component deterioration or failure due to exposure to nuclear radiation. An external loop of tungsten wire capable of direct radiation into space, will be used for the parasitic load.

2.1.2 Conceptual System Design

Three conceptual vehicle designs are described. The basic hydrogen configuration contains a 3700 lb. hydrogen storage capacity located within the Snap 8 radiator cavity. Overall vehicle length is 35 feet. The shared-tank configuration utilizes an excess 2100 lb. hydrogen storage capacity within the Centaur stage to achieve a reduction in vehicle length to 28 feet. Storage requirements within the arc jet vehicle are reduced to 1900 lbs. The ammonia configuration is identical in external dimensions to the shared-tank configuration but must contain the full propellant capacity. This requirement will result in a higher vehicle center of gravity for the ammonia vehicle than for the hydrogen shared-tank vehicle.

2.1.3 Engine Performance

Hydrogen engine performance up to .75 lbs. of thrust at 1000 seconds specific impulse and up to 1100 seconds at .5 lbs. of thrust can be achieved with a 30 KW engine configuration. The maximum ammonia engine performance that can be achieved with .5 lbs. of thrust is a specific impulse of about 800 seconds.

2.1.4 Mission Analysis

A payload capability of 2100 lbs. in excess of the 30 KW Snap 8 power system can be achieved by the current XT-761 arc jet engine in transferring from a 300 nautical mile parking orbit to a 24 hour equatorial orbit. Payload capabilities for transfer to a 20 mile altitude orbit about the moon are 880 lbs. in excess of the Snap 8 power system. A satellite network distribution capability for spacing ten satellites at equally spaced intervals about a 6000 mile altitude orbit can be achieved by the XT-761 arc jet engine. Ten 354 lb. satellites can be distributed if the launching is achieved by the Centaur booster and ten 1055 lb. satellites if launched by the Saturn C-1 booster.

Superior mission performance capabilities have been obtained with hydrogen propellant within the minimum current arc jet engine specifications of .5 lbs. of thrust and 1000 seconds specific impulse. In addition, hydrogen operation offers the greatest potential payload improvement on the basis of maximum specific impulse attainable.

2.1.5 Space Program Applications

The most attractive applications identified for the 30 KW arc jet propulsion system are the Advent, Relay II, and the commercial communication satellite programs currently in the development or planning phases. If selected for one or more of these programs additional possible applications are Aeros and Astrostat. For each of these missions, the arc jet propulsion system provides a capability for achieving substantial increases in payload capabilities beyond those possible with an all-chemical boost system.

2.2 Conclusions and Recommendations

The following conclusions and recommendations are submitted as a result of the above studies:

1) The 30 KW Arc Jet Engine can provide substantial improvements in payload capabilities for the Advent, Relay II, and other operational communication satellite systems utilizing a 24 hour equatorial orbit. It is recommended, therefore, that the engine be considered for the final boost phase for the current systems and as a means for achieving system growth capabilities.

2) The 30 KW Arc Jet Engine can provide substantial reductions in the number of launchings required for the establishment of the commercial communication satellite, Midas, and Samos satellite networks. It is recommended, therefore, that the engine be considered for this satellite distribution function as a means for minimizing launch vehicle requirements for establishing and maintaining such operational satellite networks.

3) If an arc jet vehicle is developed for (1) it can be utilized for the Aeros and Astrostat programs, as well.

4) Suitably attractive mission performance capabilities can be achieved with either hydrogen or ammonia as the engine propellant. For those applications in which the premium is on maximizing payload capabilities, hydrogen should be utilized at the maximum specific impulse level compatible with the available state-of-the-art. For those applications requiring extended periods of either continuous or intermittent propulsion system operation, ammonia should be utilized as the propellant in order to minimize cryogenic storage requirements.

5) Storage volume requirements for hydrogen operation will compromise arc jet vehicle design. This problem can be minimized - with no sacrifice in performance - by utilizing excess hydrogen storage capacity available in the Centaur stage.

6) Additional system components will be required in order to transmit the power output of the Snap 8 to the Arc Jet Engine and to obtain stable arc operation. The most significant component required for this power transmission function will be a three-phase transformer with built-in inductance.

7) Propellant feed to the engine can be accomplished by controlled boil-off of the cryogenic propellant storage system.

3. SYSTEM COMPONENTS

The Snap 8 power output specifications and the 30 KW Arc Jet power and propellant input requirements have been utilized to identify and define mechanical and electrical system components needed for operation as a complete propulsion system. Additional components which effect the design or operation of the engine have also been considered. The effect of the space environment and the mission requirements for each of the selected candidate missions has been considered whenever possible.

3.1 Snap 8 Power System

The Snap 8 Power System consists of a liquid metal cooled nuclear reactor, turbine, radiator, generator, and other auxiliary equipment. It provides an electrical power output of 30 KW of three phase, 1000 cps, alternating current at a regulated line to line voltage of 75 volts RMS. A growth version of the basic system will utilize a single reactor with twin turbines, radiators, generators, etc. to provide an electrical power output of 60 KW. Estimated weight for the 30 KW configuration, including shadow shielding for the equipment, is 2000 lbs; and for the 60 KW configuration, is 3000 lbs.

3.2 30 KW Arc Jet Propulsion System

NASA is currently sponsoring programs for the development of 30 KW arc jet propulsion systems. The XT-761 engine is the 30 KW thermal arc engine being developed by the General Electric Company.

The engine requires 30 KW of three phase AC power at a frequency of 1000 cps and a phase to phase voltage of between 300 and 450 volts RMS. The engine uses hydrogen as a propellant and is regeneratively cooled by a double pass heat exchanger. Successful operation of the engine has been achieved with ammonia as the propellant. The propellant is heated by arcs established between each possible pair of the three electrodes (delta connection). Figure 3.1 illustrates the configuration selected for the XT-761 engine.

When operated from the SNAP 8 power supply, the XT-761 engine requires a step-up transformer to obtain the desired operating voltage and series impedances in each of the power leads to provide stability to the arcs and to set the desired operating point. Starting and shutdown procedures for the engine require switchgear and a starter circuit.

The thrust chamber is cylindrical with a diameter of 3.5 inches and a length of 7 inches. The chamber weighs 4 pounds.

ITEM	IDENTIFICATION NO.	DESCRIPTION OR NAME	ZONE	GROUP NO. & QTY 1	3	5	7
				X	X	X	X
1		ASSY					
2	4012080-902P1	HOUSING PILOT UNIT	B-8	1	1	1	1
3	4012080-907P1	FLANGE PILOT UNIT	B-8	1	1	1	1
4	COML STN STL	BOLT HEX HD THD #10(.190)32NF-2A 1.000 LG	B-9	6	6	6	6
5	COML STN STL	WASHER PLAIN 3/16 ID - 7/16 OD X .049 THK	B-9	15	15	15	15
6	U-2105-00375 HP	O RING A	E-7	4	4	4	4
7	4012080-917G1	STUD PILOT UNIT	J-10	1	1	1	1
8	4012080-912P1	WASHER	C-7	6	6	6	6
9	4012080-913P1	STUD PILOT UNIT	J-10	3	3	3	3
10	COML STN STL	NUT HEX THD #10(.190)32 NF-2B	K-10	7	7	7	7
11	4012080-921P1	STUD PILOT UNIT	E-10	1	1	1	1
12	4012080-919P1	NOZZLE PILOT UNIT	J-3	1			
13	COML-STN STL	WASHER PLAIN 3/16 ID - 7/16 OD X .046 THK.	L-8	12	12	12	12
14	COML STN STL	NUT HEX THD #8 (.164) 32 NC-2B	K-8	6	6	6	6
15	COML STN STL	WASHER PLAIN 1/4 ID - 5/8 OD X .065 THK	L-8	1	1	1	1
16	4012080-915P1	STRAP PILOT UNIT	L-8	3	3	3	3
17	COML STN STL	NUT HEX THD .250-20NC-2B	L-8	1	1	1	1
18	4012080-905P1	FLANGE PILOT UNIT	L-7	1	1	1	1
19	4012080-906P1	FLANGE PILOT UNIT	L-7	1	1	1	1
20	4012080-910P1	GASKET PILOT UNIT	K-7	1	1	1	1
21	4012080-908P1	RING PILOT UNIT	L-7	1	1	1	1
22	4012080-930G1	ELECTRODE	L-5	3	3	3	3
23	4012080-918P1	PLENUM CHAMBER PILOT UNIT	L-4	1	1	1	1
24	4012080-923P1	FLANGE PILOT UNIT	L-4	1	1		
25	U-4218-03500 SE	O RING A	E-3	2	2	2	2
26	4012080-909P1	RETAINER PILOT UNIT	C-5	1	1	1	1
27	4012080-911P1	SCREEN PILOT UNIT	C-7	6	6	6	6
28	COML STN STL	BOLT HEX HD THD #10(.190)32NF-2 X .750 LG	L-3	6	6	6	6
29	4012080-904P1	SLEEVE PILOT UNIT	C-8	1	1	1	1
30	4012080-916P1	GASKET PILOT UNIT	D-8	1	1	1	1
31	4012080-914P1	COVER PILOT UNIT	D-8	1	1	1	1
32	4012080-911P2	SCREEN PILOT UNIT	D-8	6	6	6	6
33	4012080-923P2	FLANGE PILOT UNIT	L-4			1	1
34	4012080-920P2	NOZZLE PILOT UNIT	J-3				1
35	4012080-919P2	NOZZLE PILOT UNIT	J-3		1		
36	4012080-920P1	NOZZLE PILOT UNIT	J-3			1	

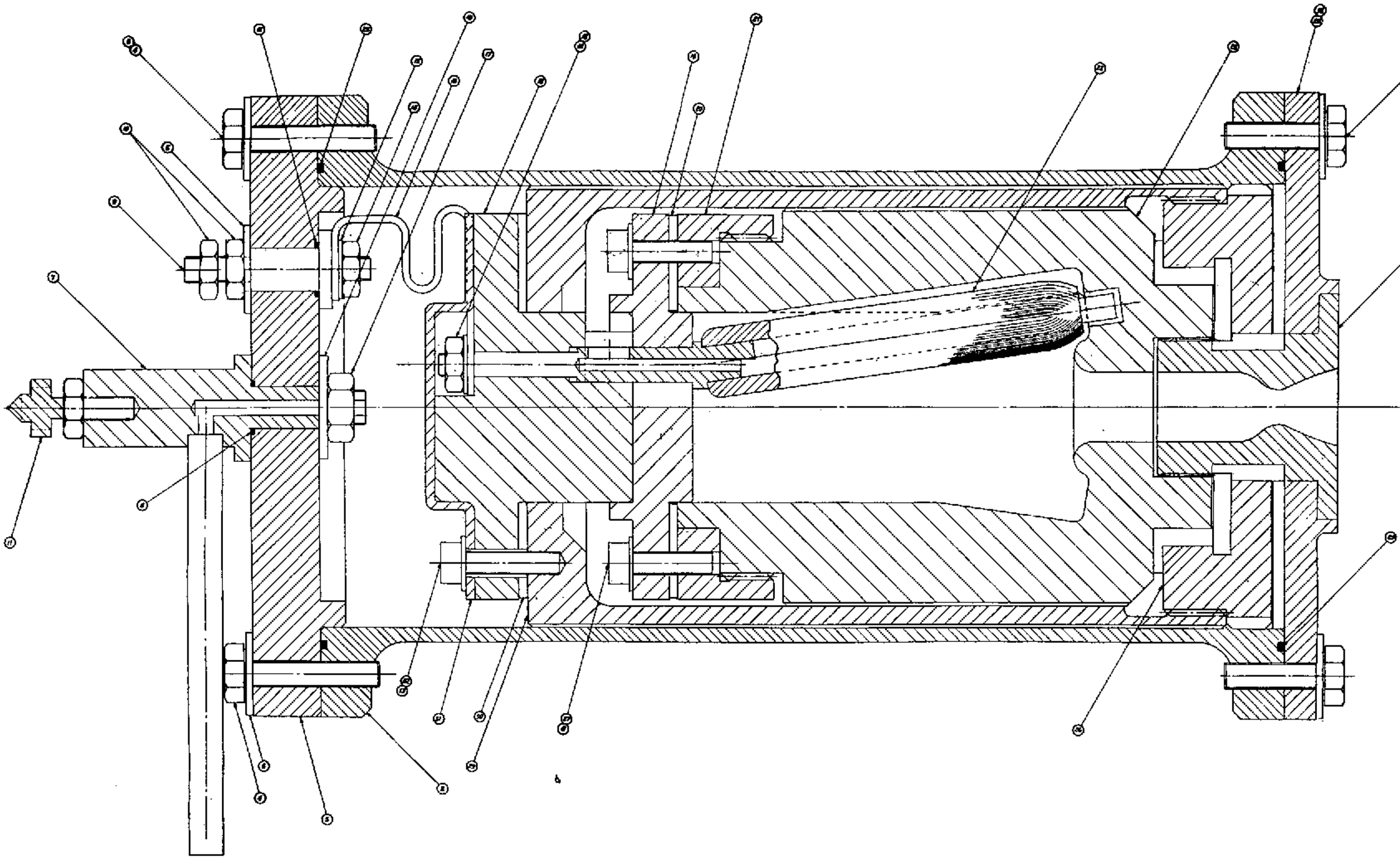

Figure 3.1 Pilot Unit Assembly.

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3.3 Mechanical System Components

The major mechanical system components investigated include the propellant storage, propellant feed, and thrust vectoring systems. In the following, the major design considerations will be discussed and the selected solutions indicated.

3.3.1 Propellant Storage System

Liquid hydrogen, helium, and ammonia were considered as potential engine propellants. Liquid hydrogen and helium are cryogenic fluids which pose special handling and storage problems. In the following sections, the physical properties affecting storage system design will be indicated and the resulting design obtained will be discussed.

3.3.1.1 Propellant Properties

The most important fluid property of the fluids considered will be specific volume. Table 3.1 lists typical properties of each of the propellants.

TABLE 3.1

PROPELLANT STORAGE PROPERTIES

Propellant		Hydrogen	Ammonia	Helium
Boiling Point	^{o}F	-410	40	-452
Storage Pressure	Atm.	5	5	1
Specific Volume	$ft.^3/lb.$	.265	.025	.131

Engine operation is expected to be at 3 to 4 atmospheres chamber pressure. Hydrogen and Ammonia storage properties are, therefore, shown at 5 atmospheres pressure in order to provide some additional pressure for flow metering into the engine. Helium data are shown at 1 atmosphere since storage at 5 atmospheres would be above the critical point of Helium. Figure 3.2 illustrates the storage volume requirements as a function of propellant weight.

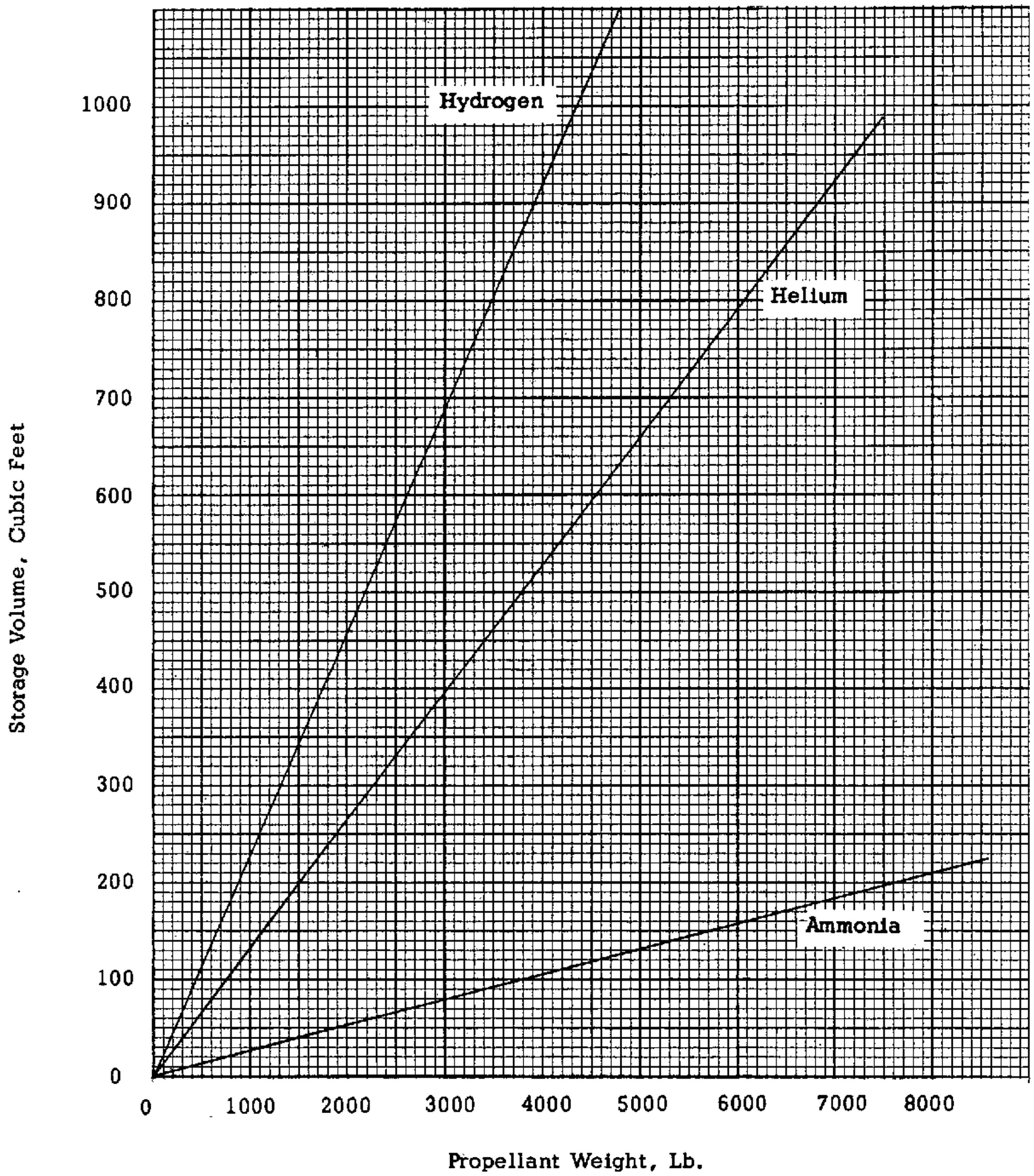

Figure 3. 2 Storage Volume Requirements for Liquid Propellants Considered.

Although the subsequent discussion considers each of the fluids, primary attention has been focused on the utilization of hydrogen because of its apparent superior engine performance and life capabilities.

3.3.1.2 Storage System Design

The resulting storage system design is shown in Figure 3.3. A double-wall construction has been utilized. The inner shell utilizes a .050 inch aluminum alloy wall. Estimated tank weight-including flanges, vacuum provisions while on the launch pad, and the necessary valves and piping accessories - are illustrated on the lowest curve of Figure 3.4. The second curve shows the tank weight with a .005 inch sheath of stainless steel spaced 3/8 inch away from the aluminum alloy wall. This sheath will act as a radiation heat shield in space as well as a bumper shield for meteorite protection.

The highest curve shows the tank weight with insulation installed between the stainless steel sheath and the aluminum alloy tank. This insulation is particularly useful during the launching period in reducing the rapid boil off of hydrogen while in the atmosphere. The insulation used, Linde SI-4, has an apparent thermal conductivity between 300 and 90°K of $2.5(10)^{-5}$ BTU/hr. ft. °F. and a density of 4.7 lbs/ft^3. During the launch preparation period, the tank would be surrounded by a ground support clam-shell which would maintain the insulation around the tank under evacuation. Just prior to launch the vacuum would be broken with a helium gas purge to prevent icing of the insulation.

The calculated heat loss rates for a tank storing 7360 lbs. of liquid hydrogen sheathed with 3/8 inch of the above insulation have been obtained from Reference [1]. These data are repeated in Table 3.2.

TABLE 3.2

HEAT LOSS RATES

Heat Flux	
Evacuated (sea level)	452 BTU/hr.
Helium Purge	800,000 BTU/hr.
In Space	237 BTU/hr.
Evaporation Rate	
Evacuated (sea level)	.76% per day
In Space (360°R skin)	.40% per day

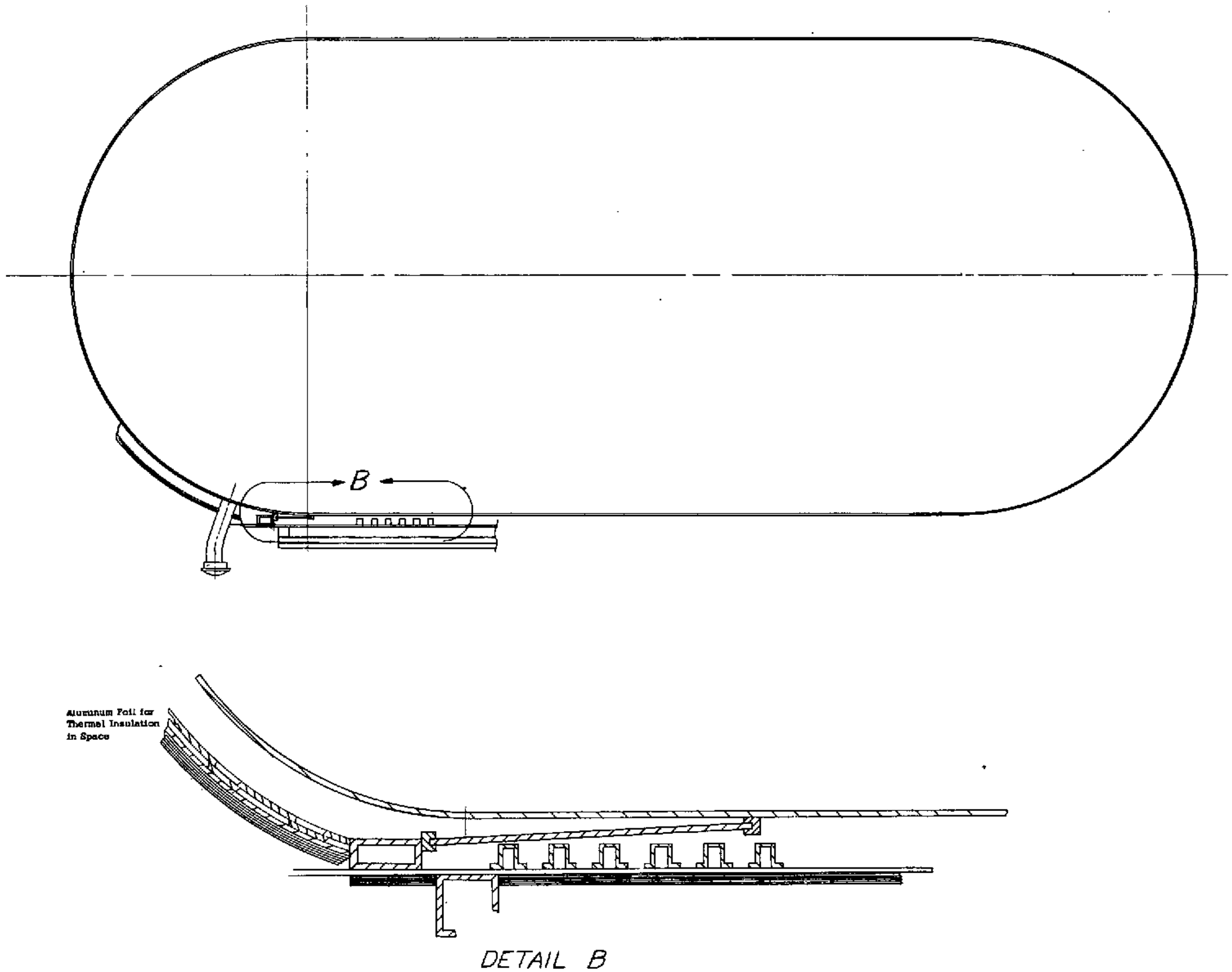

Figure 3.3 Double-Walled Hydrogen Storage System Design

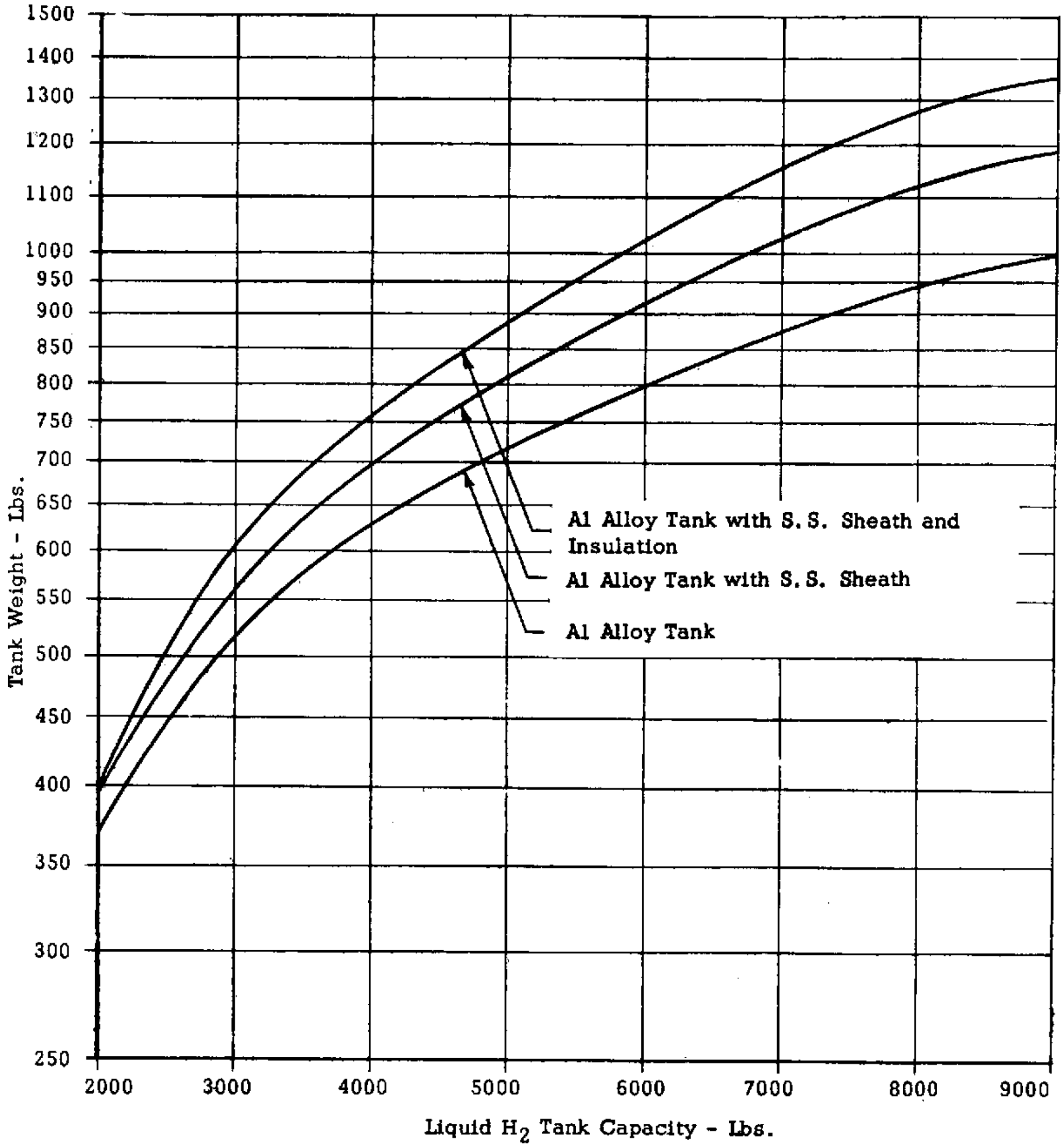

Figure 3.4 Propellant Storage Requirements for Liquid Hydrogen.

The above rates will vary with tank size, the ratio of tank volume to area, and the temperature "seen" by the tank in space. Depending upon the location of the propellant storage system relative to the Snap 8 power system, the temperature "seen" may be as high as 1800°R.

The meteorite protection afforded by this design is illustrated in Figure 3.5. This curve contains the limiting mission duration for a 90% probability of completing the mission without a meteorite penetration. The methods of R. L. Bjork[2] and the correlations of M. Kornhauser[3] were used. Data are shown as a function of arc jet engine thrust and specific impulse for both 30 KW and 60 KW configurations. Mission times in excess of those shown will require increased tank dimensions and weight for the same degree of meteorite protection.

An alternate storage system design investigated utilizes two .020 inch aluminum alloy walls separated by a 3/8 inch vacuum for insulation while in the atmosphere. Space insulation is provided by multi-layer thicknesses of aluminum foil which are wrapped around the outer wall. Adequate meteorite protection is afforded by the double-wall construction. This design will provide somewhat better insulation in space but will be somewhat heavier.

The number of foil layers required for limiting propellant boil-off rate in space to the value required for engine feed has been investigated for both the basic hydrogen vehicle configuration and for the shared-tank vehicle configuration. Calculations have been based on the following assumptions:

a. The total radiation incident to the tank is due to solar radiation at the earth's orbit plus the albedo of the earth. A value of 500 $Btu/hr.ft^2$ was utilized.

b. There is no conductive or convective heat transfer in the foil layers in space.

c. The outer surface of the external foil layer is coated in order to achieve a high black body emissivity with a low absorption coefficient to the incident radiation. An absorption coefficient of .05 and an emissivity of .8 were utilized for the outer surface.

d. The surface of the remaining foil layers are identical. An emissivity of .01 and a foil thickness of .0001 inches were used for each layer.

The basic vehicle configuration would require 7.2 layers of foil to maintain the desired propellant feed rate. Eight layers would, therefore, be utilized in order to provide positive control of the engine propellant feed rate. The total foil weight required would be about 7 lbs.

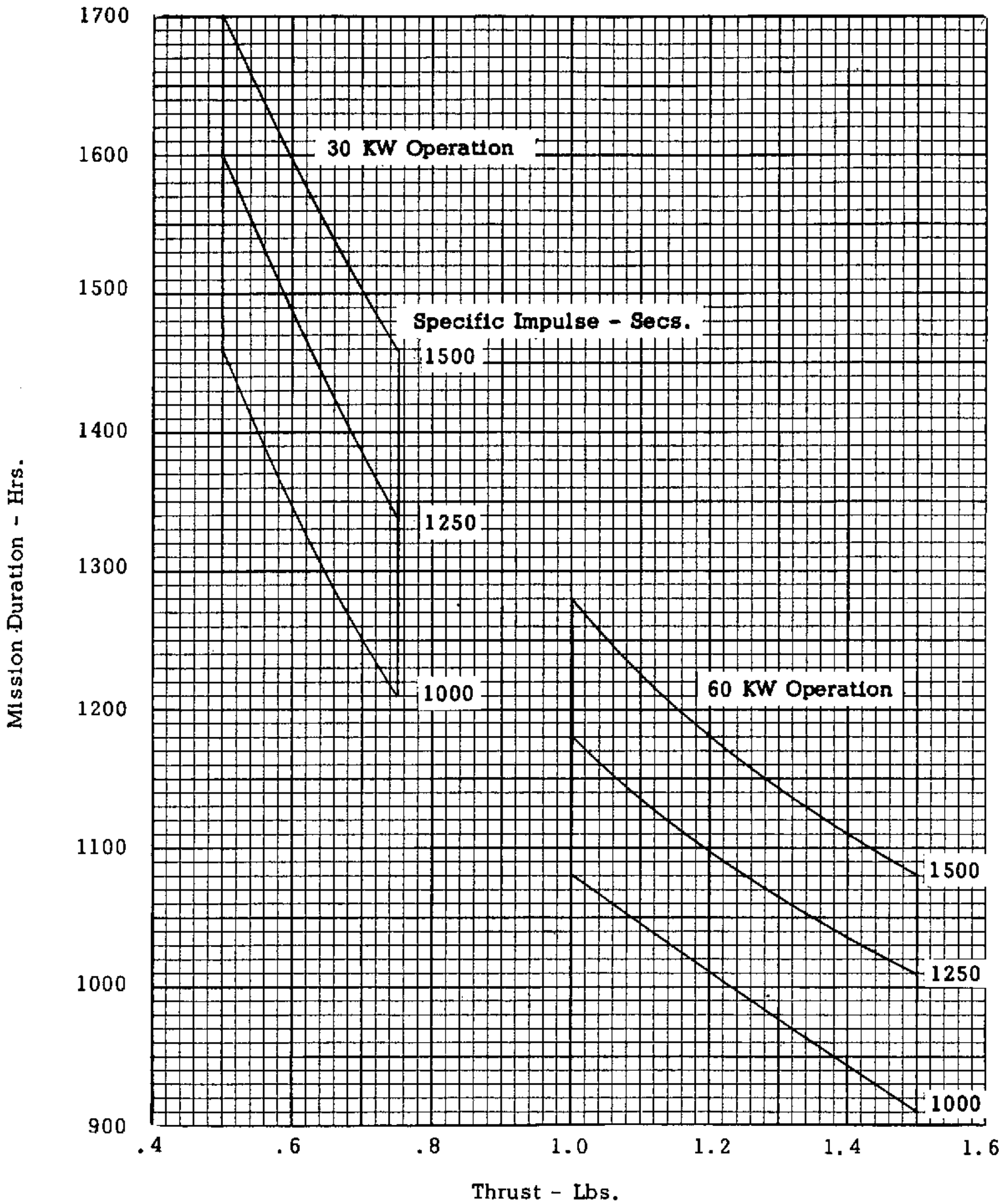

Figure 3.5 Mission Duration Limits for 90% Probability of Completion Without Meteorite Penetration.

The shared-tank vehicle configuration will require insulation about the basic Centaur stage in addition to insulation about the propellant tank in the Arc Jet stage. During the initial propulsion period in which the engine is supplied by boil-off of the Centaur tank, it is desirable to provide sufficient insulation for the final tank to achieve a negligible boil-off rate. This, however, can not be achieved with a reasonable number of foil layers.

It is recommended, therefore, that the boil-off of the final propellant tank be added to the boil-off from the Centaur tank and the total flow fed to the engine. This can be achieved by the use of 19 foil layers about each tank which will result in a combined boil-off rate that will be adequate for the engine. The Centaur tank will provide 75% of the total flow and the final stage tank will provide 25%. After the Centaur tank has been emptied and the final stage tank is being utilized to provide the complete propellant flow, its insulation could be reduced to 4 layers. The extra 15 layers could be ejected along with the Centaur stage or the deficiency in heat input to the tank provided by an electrical heater. The foil weight required would be about 24 pounds for the Centaur tank and 8 pounds for the final stage tank.

3.3.2 Propellant Feed System

The components of the propellant feed system are shown in Figure 3.6. The system consists of the pressure sensory control amplifier, heater, metering orifice, and feed lines.

The flow control measures storage pressure, compares it with a reference value, and utilizes the error signal to modulate the heat addition to the propellant within the tank-thus maintaining the desired inlet conditions to the metering orifice. The pressure sensor must be capable of operating at the cryogenic storage temperature of hydrogen. The signal required from the sensor is small. Suitable pressure sensors using differential transformers are commercially available and present no new design problems for this application.

Magnetic amplifiers using radiation tolerant computer diodes will be used to perform the control computation. The gas will be heated by ohmic heating in order to provide the specified propellant flow rate.

The metering orifice is located in close proximity to the propellant tank. Flow regulation is provided by this orifice by maintaining required inlet conditions, i.e., temperature and pressure.

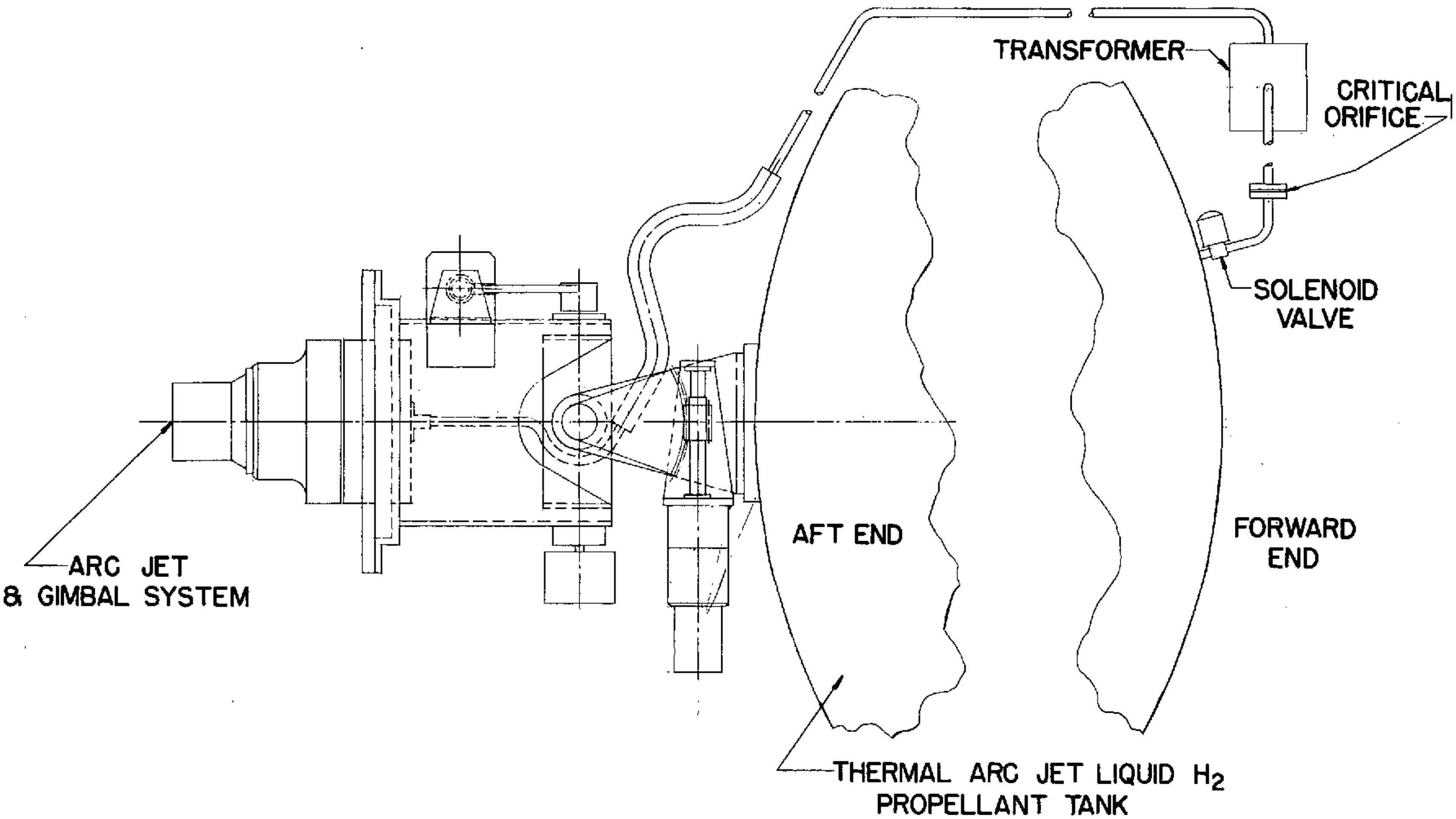

Figure 3.6 Propellant Feed System

The propellant feed lines are copper tubes which also act as electrical leads. The propellant enters the leads at the power transformer, and flows through the leads to the engine. Both engine and transformer utilize electrical insulators with a hollow conductor for electrical feed-through and propellant flow.

The propellant feed line contains an on-off valve for controlling the propellant flow. The valve is solenoid operated.

Starting requires low propellant flows in order to obtain chamber pressures suitable for establishing the arc. A second solenoid operated valve and metering orifice is therefore required in order to provide the propellant flow conditions for starting.

3.3.3 Thrust Vectoring System

Thrust vectoring is required in order to properly orient the engine thrust with respect to the path of the vehicle in order to achieve the desired orbital change and for vehicle attitude stabilization. The thrust vectoring mechanism is shown in Figure 3.7. It is mounted directly to the engine and is suspended from brackets attached to the vehicle. The vectoring mechanism consists of a gimbal ring positioned by two servomotors; one operating in the x-plane and the other in the y-plane. Motion in the x-plane is provided by the motor attached to the inner gimbal ring; and y-plane motion is provided by the motor attached to the outer ring. Thus the engine can be controlled in any position within a cone of a specified maximum apex angle.

Each servomotor is part of an integral drive assembly which also includes a tachometer generator for velocity feedback to the control, and a gearhead which provides the proper gear ratio to obtain the desired torque and speed of response. The output of the motor-generator assembly drives a bevel gear train which imparts rotational motion to each ring with respect to its shaft. A rotary position transducer is attached to each shaft opposite each motor-generator to provide position feedback to the control.

The control system for the thrust vectoring mechanism is shown in block diagram form in Figure 3.8. It consists of two high performance position servo systems. Electrical signals proportional to the desired direction of thrust are the input to the position systems. Position error, the difference between the desired position and the position indicated by the feedback transducer, is amplified to operate the small servomotor. The system also includes provision for tachometer feedback to obtain desired transient response characteristics.

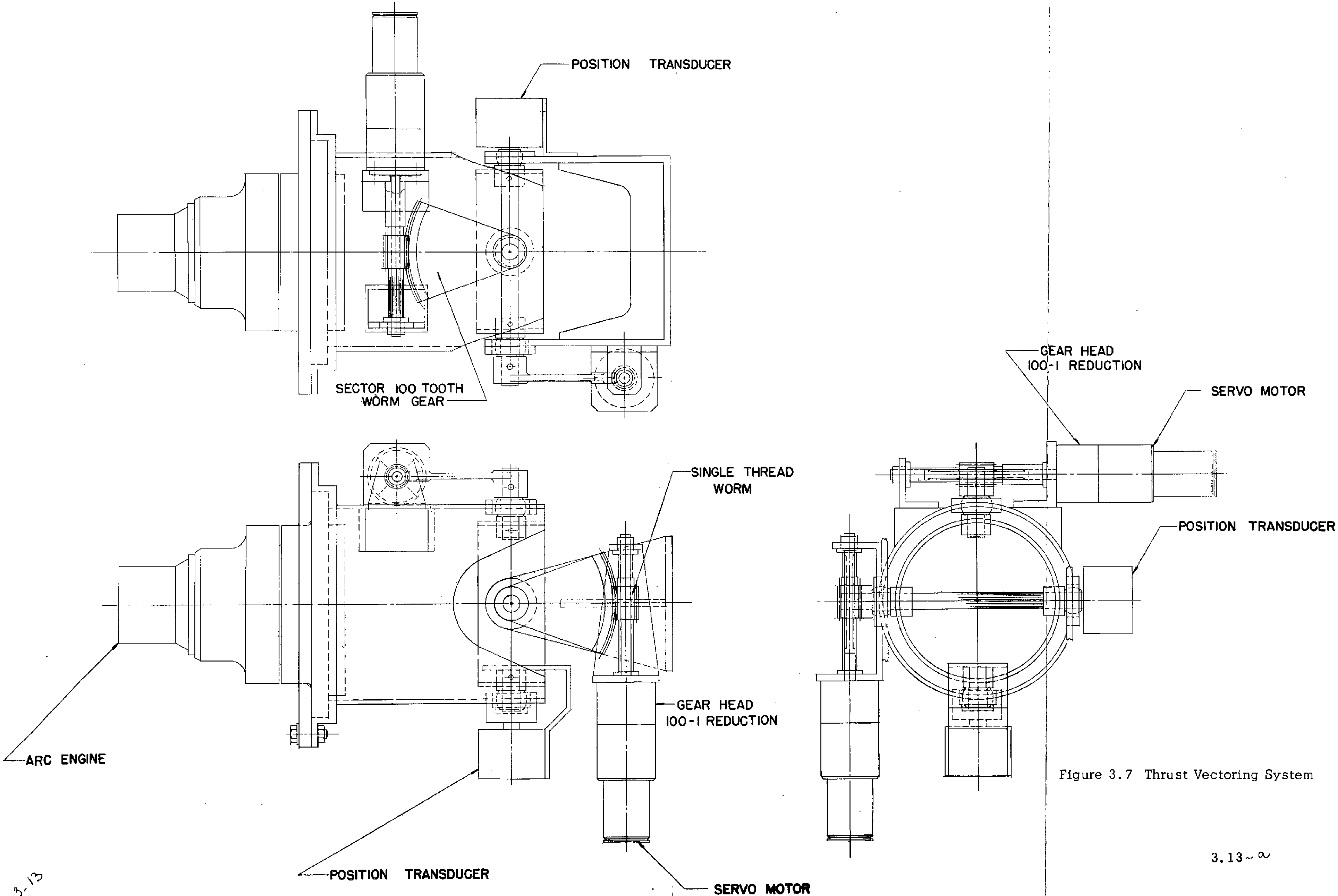

Figure 3.7 Thrust Vectoring System

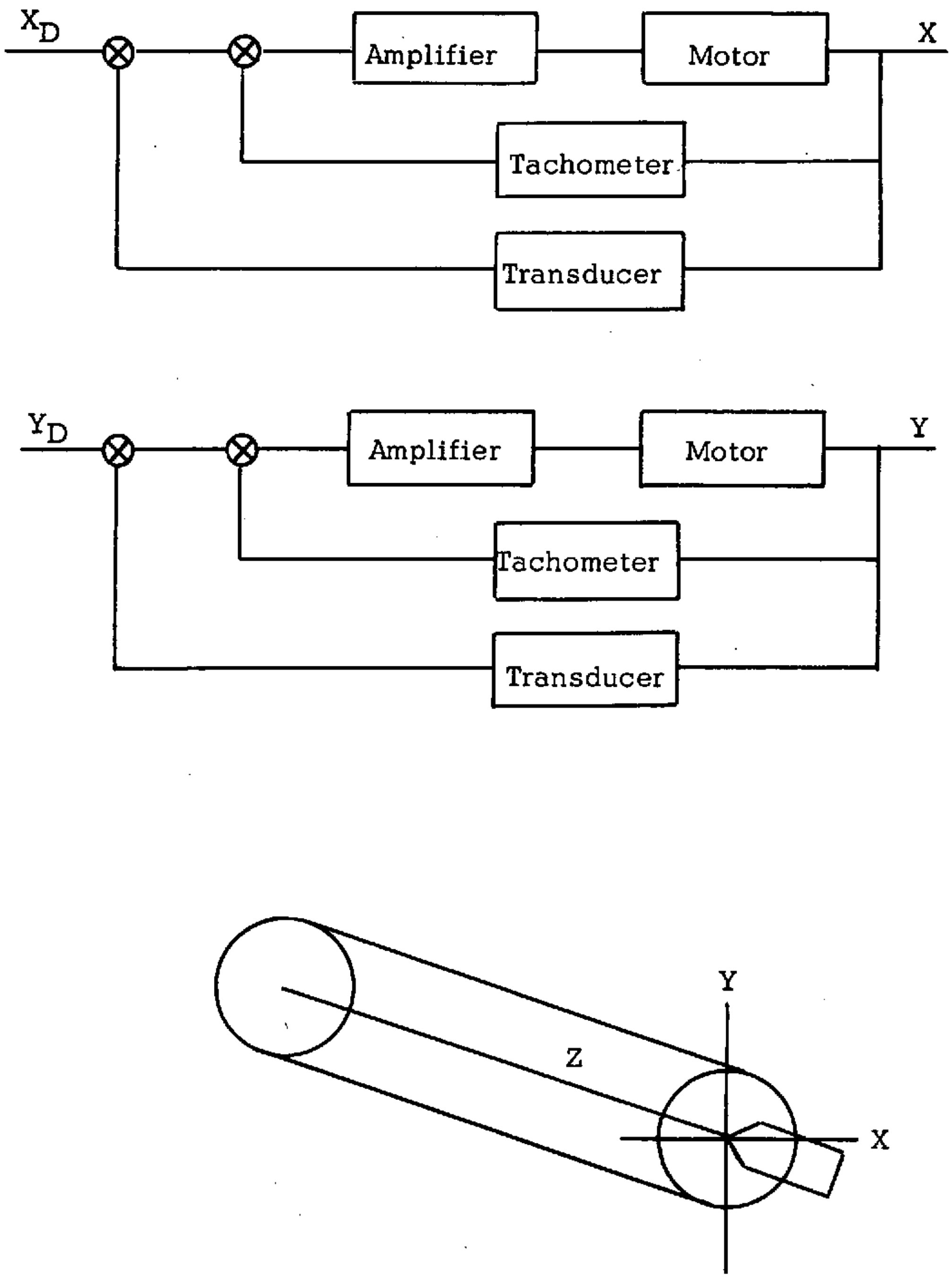

Figure 3.8 Thrust Vectoring Control System Block Diagram

Stabilization is required about all three axes of the vehicle. In a single engine - 30 KW application, yaw and pitch control can be accomplished by vectoring the engine in the x and y directions. Roll being a rotation about the principle or Z axis of the vehicle cannot be controlled by vectoring the engine and a separate source of torque must be provided for moment about this axis. This can be furnished by additional small opposed pairs of arc jet units which are rigidly attached to the vehicle and are operated alternately depending on the direction in which moment is required. Use of two such pairs will produce roll torque without propulsive thrust. Arc engines of the 1 KW size should be adequate.

Attitude control for dual engine - 60 KW applications can be accomplished by vectoring the engines. Yaw control is provided by vectoring both engines equally in the x direction. Similarly pitch control can be accomplished by vectoring both engines equally in the y direction. Thus pitch and yaw control is obtained by orientation of the thrust vector with respect to the vehicle axis. Roll control may be provided by the differential position of the engines in the y direction. This differential positioning of the engines provides a moment about the z-axis of the vehicle without interferring with normal pitch control.

The control amplifier will be built using magnetic amplifiers utilizing radiation resistance diodes. Such diodes will be either solid state devices or ceramic vacuum tubes. Research is in progress in both areas to develop radiation tolerant computer diodes.

3.4 Electrical System Components

The electrical system components consist of a transformer to obtain the required engine operating voltage, inductance to provide stabilization and to set the operating point, transmission lines, switchgear for turning the engine on and off, and a resistance ballast or parasitic load to be used to radiate excess energy from the power supply when the engine is turned off. A block diagram of the complete system is illustrated in Figure 3.9.

3.4.1 Transformer and Inductors

In order to match the engine requirements to the SNAP 8 power supply specifications, a transformer is required to step-up the voltage and to reduce the current input to the engine. The transformer will be a step up three phase wye connected auto-transformer with the output connected in delta. The design of this transformer will present problems of insulation and cooling.

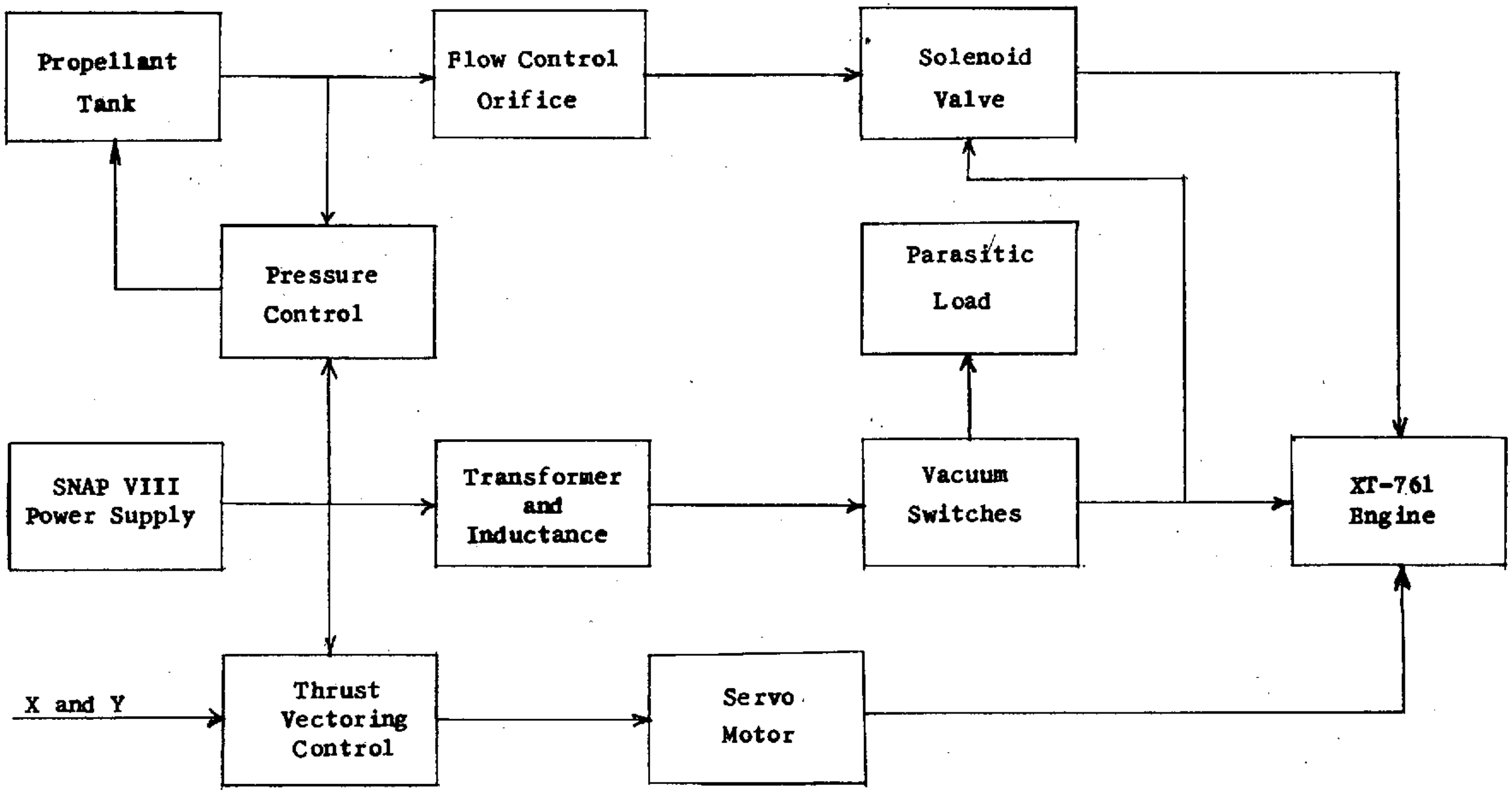

Figure 3.9 Electrical System Block Diagram

The insulation problem is the prevention of arcing and corona discharges in the space environment at the high voltages required. One possible solution involves the use of a hermetically sealed transformer that does not see the high vacuum environment.

Cooling is required in order to remove the heat energy generated by both the copper and the iron losses in the transformer. Total losses can be kept small and the greatest cycle efficiency obtained by cooling the transformer with the propellant. One means for regenerative cooling of the transformer is the use of hollow tubes for the conductor with a portion of the propellant flow passing through each winding. The propellant would enter the transformer at the junction of the wye which is at ground potential and exit at the high voltage transformer taps. The weight of such a 30 KW transformer and associated power leads would be about 90 lbs. The transformer should be physically located as close to the alternator as is feasible in order to permit transmission of the power at low currents thus reducing line losses and/or the weight of leads.

Design of stabilizing inductances presents the same problems of insulation and cooling as does the transormer, and the same approach to their solution is suggested. The inductances may be separate components wound on toroidal iron cores or may be provided by leakage reactance designed into the transformer.

Figure 3.10 presents a preliminary design of such a propellant cooled autotransformer. The salient features of this transformer are the hermetic sealing of the transformer to prevent arcing and corona discharge between the windings, spacing of the windings on the core to provide sufficient leakage reactance to stabilize the engine, and cooling of the transformer by passage of the propellant over the windings. In this preliminary design, the propellant enters the transformer through the propellant case and exits through the high voltage electrical power leads.

3.4.2 Electrical Switchgear and Parasitic Load

Electrical switchgear and a parasitic load are required for turning the engine on and off without introducing load distrubances to the power supply.

The first approach considered to meet the switching requirements was the use of solid state devices to take advantage of their low weight and absence moving parts. However, the radiation environment seen by these components will not permit their use. Vacuum switches are currently available, however, which are capable of switching currents of this magnitude.

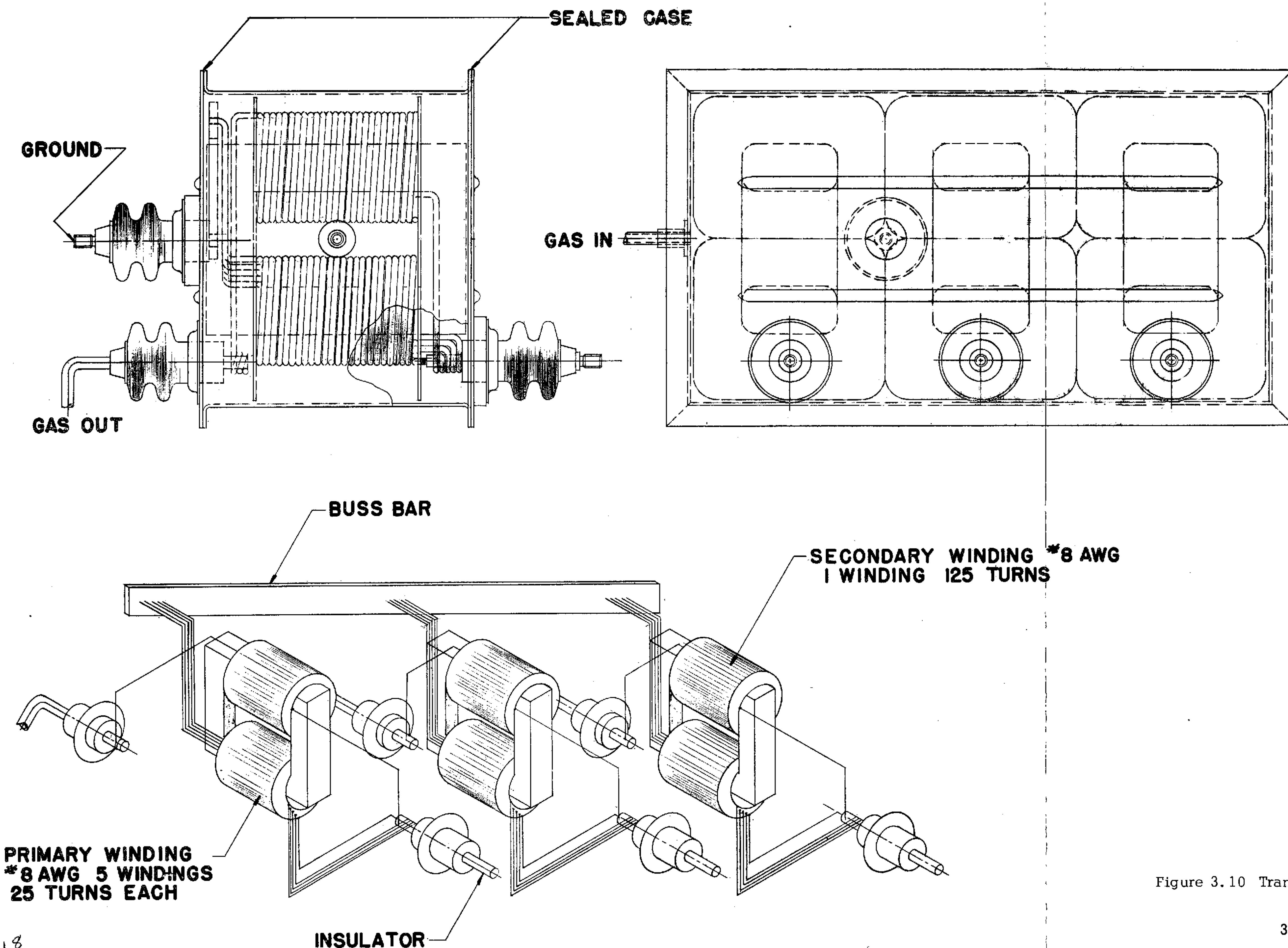

Figure 3.10 Transformer

3.18

Make-before-break switches are recommended. The switches are to be located after the transformer thus permitting the parasitic load to operate from the higher voltage and reduced currents available.

The parasitic load will consist of a loop of tungsten wire located along the outer surface of the vehicle and radiation shielded from it. All power expended in this load will be lost by radiation.

4. CONCEPTUAL SYSTEM DESIGN

Conceptual system designs have been generated in order to identify overall vehicle requirements and characteristics. The individual component designs described in the previous section have been utilized along with the latest available data on the Snap 8 and the Atlas-Centaur launch vehicle configurations.

Initially, vehicle design criteria were established. These criteria paid particular attention to the arrangement of the major components in order to maximize the volume available for payload while minimizing the height of the vehicle center of gravity above the Centaur adapter and retaining the basic Centaur outside diameter limitations.

These design criteria were then utilized to define a basic hydrogen vehicle configuration, an integrated Centaur-hydrogen vehicle, and an ammonia vehicle configuration.

4.1 Design Criteria

The following design criteria were established:

(1) The vehicle shape should be either conical or cylindrical with a maximum diameter no larger than the outside diameter of the Centaur launch vehicle.

(2) The vehicle length should be as close as possible to a minimum length needed to contain the Snap 8 radiator in its stored position.

(3) The cavity formed by the radiator in its stored position should be utilized to contain the propellant tank and the payload.

(4) The propellant tank should be placed between the payload and the reactor in order to minimize radiation exposure of the payload.

(5) The major components should be arranged - within the above restrictions - to minimize the vehicle center of gravity above the Centaur adapter. Failure to do this will result in reduced Centaur orbital payload capabilities.

4.2 Basic Hydrogen Vehicle Configuration

A basic hydrogen vehicle configuration is shown in Figure 4.1 mounted on top of the Centaur launch vehicle. The vehicle is cylindrical with a tapered nose cone. The diameter of the cylindrical section is 10 feet and the length is 24 feet. The nose cone length is 11 feet.

The Snap 8 power system utilizes two eight foot diameter rings which are located 15 feet apart. The radiator wings are wrapped around these rings during launch and extended for operation in space. The generator and turbomachinery are supported by the top ring and extended into the vehicle nose cone. The reactor and shadow shielding are also attached to the top ring and extended into the nose cone. The overall Snap 8 system length is 23 feet.

A double-walled hydrogen storage tank is shown between the two Snap 8 support rings and within the radiator cavity. Its outer wall is extended to the base of the nose cone and used to support the Snap 8 system. The lower end of the tank is supported from a truncated cone which is attached to the Centaur at the payload separation line indicated. The space between the two walls is evacuated in order to provide thermal insulation in the atmosphere. The double wall construction provides both insulation and meteorite protection. A multi-layer expandable aluminum foil around the outside of the tank is used to provide thermal insulation in space.

Two propellant tank capacities are shown. The 3100 lb. hydrogen capacity will be sufficient for a 24 hour transfer mission and the 3700 pound capacity for the stationary satellite transfer mission.

The payload is shown attached to the truncated cone below the propellant storage tank in order to provide a maximum separation distance from the reactor. The propellant provides additional radiation shielding for the payload. Approximately 300 cubic feet are available for payload storage.

The arc jet engine, controls and thrust vectoring system are integrated into a relatively small compact package, less than one cubic foot in volume. This package is mounted to the bottom of the propellant tank or, as an option, can be mounted on telescoping rods which would extend the engine several feet away from the tank after stage separation in order to provide additional moment arm for thrust vectoring. The transformer is located close to the Snap 8 alternator.

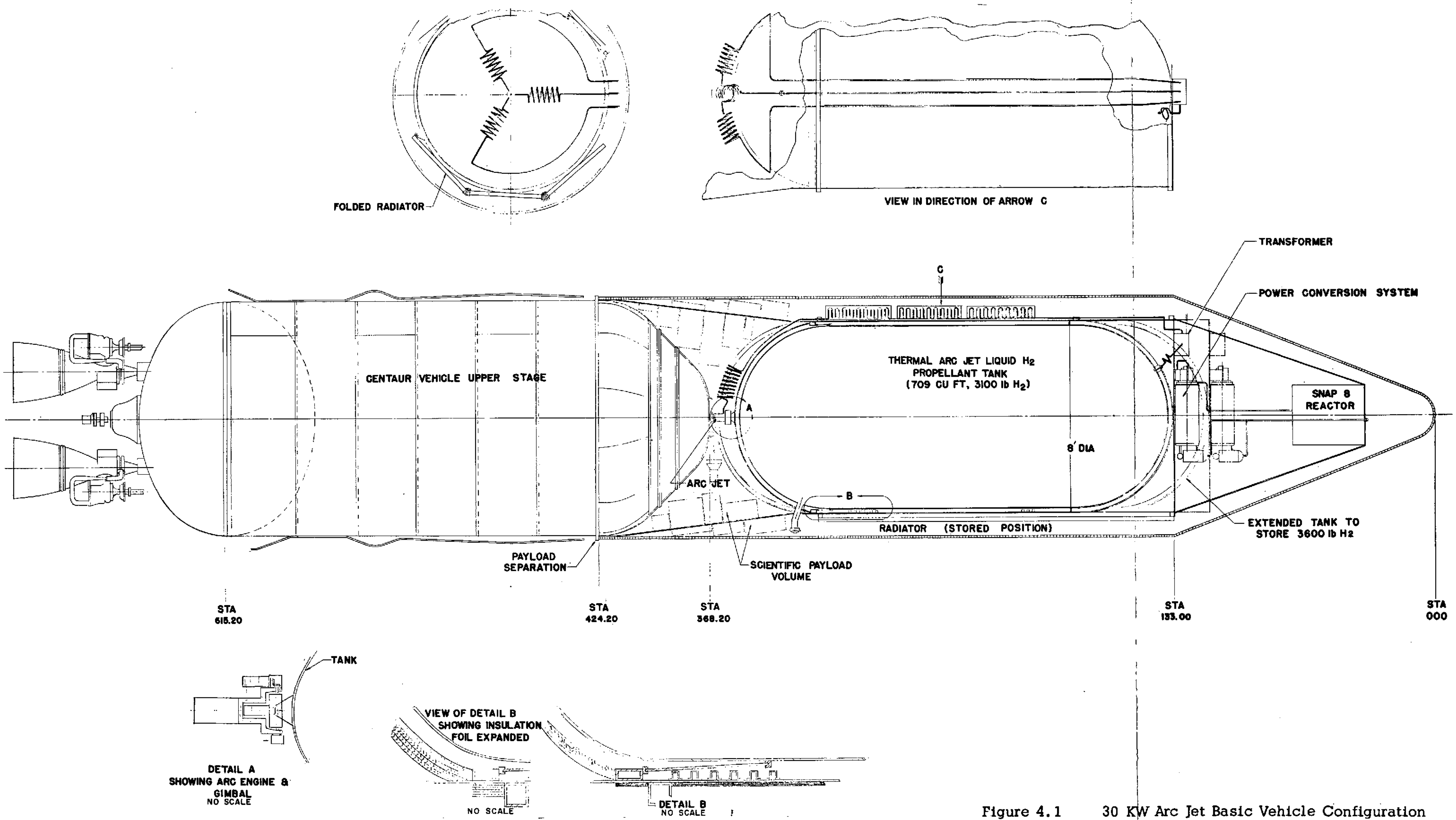

Figure 4.1 30 KW Arc Jet Basic Vehicle Configuration

The propellant fuel lines extend from the top of the storage tank to the transformer and from there extend down the side of the tank to the engine. The power transmission lines are used to enclose the propellant feed lines in order to reduce I^2R losses.

4.3 Shared-Tank Configuration

A shorter hydrogen vehicle configuration with a lower center of gravity can be obtained by integration with the Centaur vehicle. The Centaur stage is a multi-purpose vehicle which is designed for use with both Atlas and Saturn lower stages. When used for establishing an 8500 lb. payload in a 300 nautical mile orbit, the stage will have an excess hydrogen storage capacity of 2100 lbs. The shared-tank configuration utilizes this excess capacity to store hydrogen for arc jet propulsion.

A conceptual shared-tank design is illustrated in Figure 4.2. The reduction in propellant storage requirements for the final stage permit a reduction in vehicle length to 28 feet - 7 feet less than the length of the basic configuration.

A propellant storage capacity of 560 lbs. in the final stage is illustrated. This will be sufficient for injecting the current Advent vehicle into a stationary satellite orbit. A storage capacity in excess of 400 cubic feet would be available for payload.

Payload capabilities with increased propellant storage are essenially the same as the basic configuration. The reduction in propellant tankage weight brought about by the storage of 2100 pounds of propellant in the Centaur tank is offset by slightly increased total propellant requirements to carry the additional 2000 lb. weight of the Centaur during the first orbital propulsion phase. A total propellant capacity of 4000 lbs. - with 1900 lbs. stored in the final stage vehicle - will result in identical performance to the 3700 lb. capacity basic hydrogen configuration.

The advantages of the shared tank configuration are its reduced vehicle length and lowered center of gravity which could result in a greater initial orbital weight capability for the Centaur vehicle.

Two vehicle configurations are illustrated in Figure 4.2. The first shows the payload compartment between the propellant tank and the reactor. The other arrangement shows the propellant tank in the middle. The preferred arrangement will depend on the relative density of the payload and its sensitivity to radiation exposure.

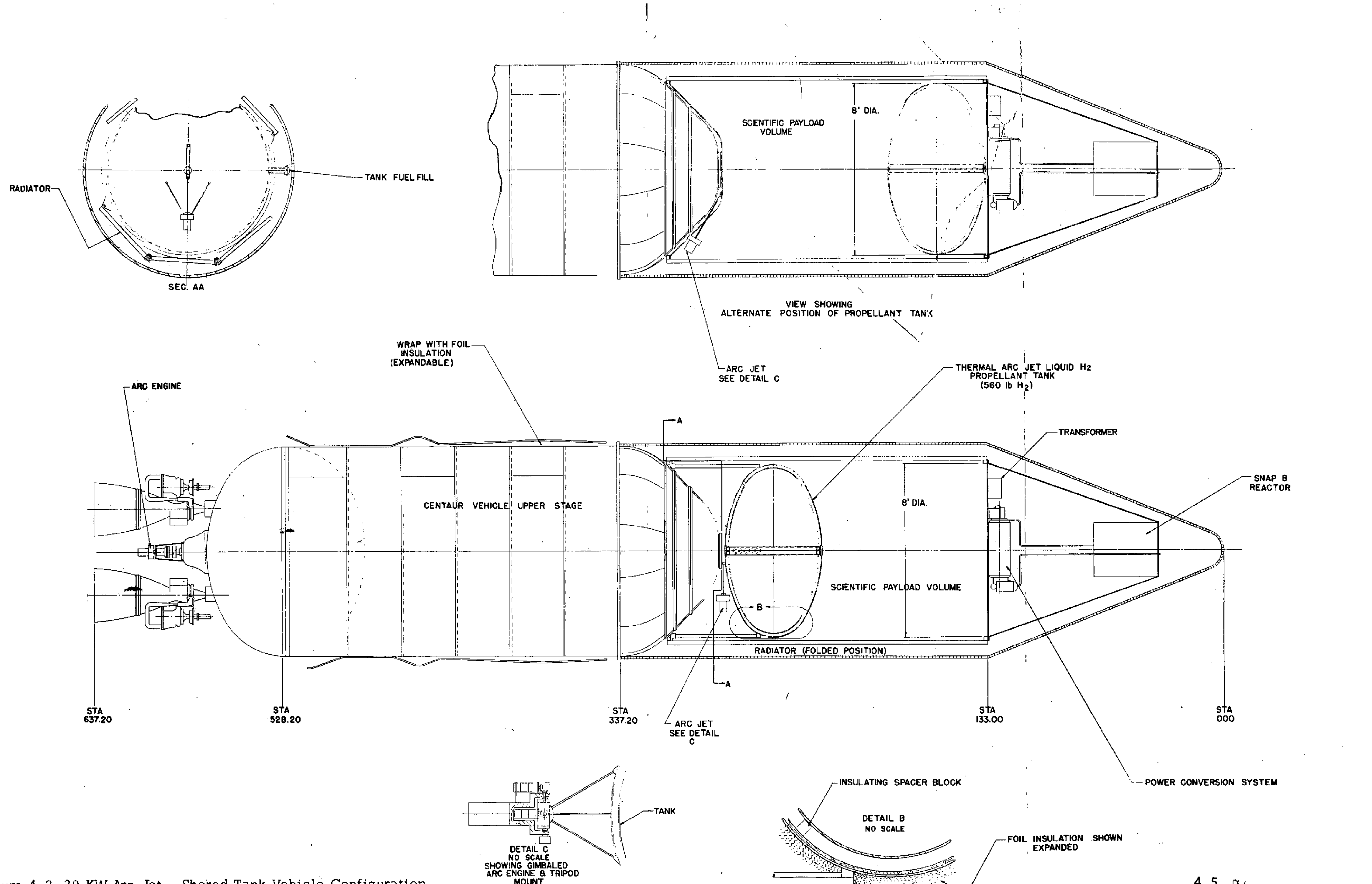

Figure 4.2 30 KW Arc-Jet - Shared Tank Vehicle Configuration

4.5

The mode of operation is illustrated in Figure 4.3. Phase A shows the complete vehicle immediately after lift-off by the Atlas booster and sustainer engines. At burn-out of the booster engines, the nose cap surrounding the Snap 8 and the booster engines are separated from the vehicle. This is shown by Phase B. Acceleration is continued with the Atlas until sustainer burn-out. This is shown by Phase C. The remainder of the Atlas stage is then separated and the Centaur ignited. The Centaur is used to establish the system in a 300 Nautical mile parking orbit with at least one intermediate coasting period.

The two LR-115 Centaur engines are then separated, any excess oxygen in the stage vented to space, and the Snap 8 radiator erected. This is illustrated as Phase D. The Snap 8 is then turned on and its power output transmitted to an arc jet engine mounted at the bottom of the Centaur stage. Propellant feed to the engine is from the Centaur storage tank until the supply is exhausted. At this point, the Centaur stage is separated from the rest of the vehicle and the power is switched to a second arc jet engine located at the bottom of the arc jet vehicle. This is indicated by Phase E. Propulsion continues in this mode until the desired orbit has been achieved.

4.4 Ammonia Configuration

An ammonia vehicle configuration will be similar in external dimensions to the shared tank configuration illustrated in Figure 4.2. The ammonia storage will, however, be contained entirely in the final stage vehicle. Operation will, therefore, be identical to that of the basic hydrogen configuration.

A spherical ammonia storage tank of 5.5 ft. diameter will be adequate for a 3700 lbs. storage capacity. Although the overall 28 foot length will be adequate for propellant and payload storage, the center of gravity for the ammonia must, of necessity, be higher than that of the shared tank arrangement. This could result in reduced initial orbital weight capabilities of the Centaur vehicle.

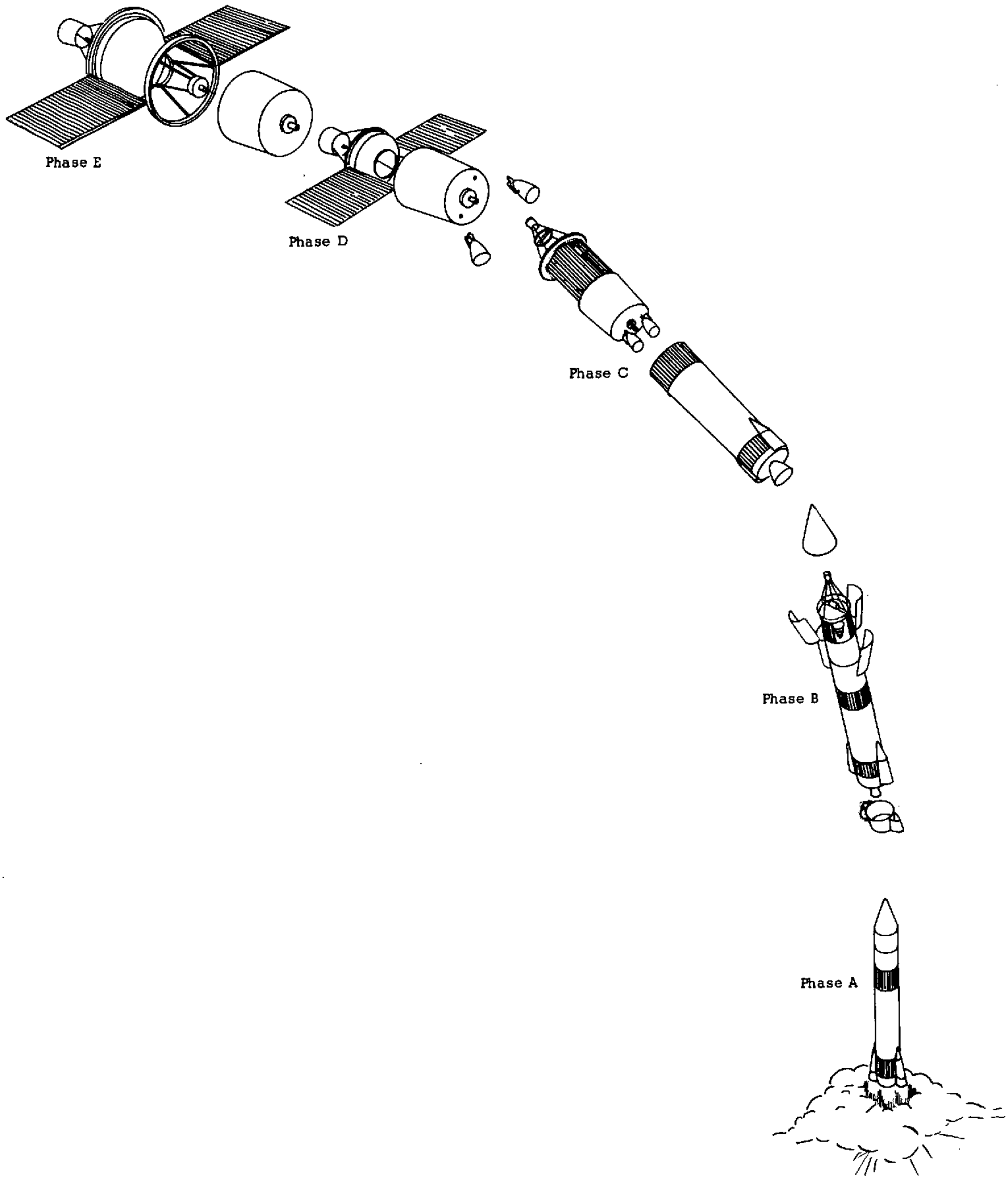

Figure 4.3 Mode of Operation for the Shared Tank Vehicle Configuration

5.0 ENGINE PERFORMANCE

The General Electric XT-761 Arc Jet Engine is currently in the twelfth and final month of its Phase 1 development program. The engine specifications established by NASA require operation at a 30 KW power input with a minimum output thrust of .5 lbs., a minimum specific impulse of 1000 seconds, and a demonstrated life capability of 50 hours operation. The engine has met or exceeded each of these specifications.

During the early phase of this study, however, very little experimental performance data was available for use in the required mission analysis studies. Consequently, thermodynamic analyses were carried out in order to generate expected engine performance characteristics for both hydrogen and ammonia propellants. These studies utilized current estimates of component efficiency levels which have since been confirmed by experimental engine operation.

5.1 Theoretical Performance

The thermodynamic analysis of the hydrogen cycle utilized an isentropic nozzle expansion process. Exhaust gas composition was determined from chemical equilibrium requirements at nozzle inlet conditions. This composition was assumed to be frozen through-out the remainder of the expansion process. [4] This assumption has been substantiated by investigations of the recombination in the nozzle using rates reported in the literature. The results of this analysis were used to generate ideal hydrogen engine performance characteristics for a 30 KW power input.

The effects of estimated component efficiencies were then used to modify the ideal performance to obtain expected engine performance characteristics. The following component efficiencies were utilized:

Electrical System Efficiency	- 96%
Nozzle Efficiency	- 92-94%
Regenerative Cooling System Efficiency	- 81-62%

The results achieved are illustrated in Figure 5.1. Data are indicated for a range of specific impulse from 1000 to 1640 seconds and for a range of chamber pressures from 1 to 10 atmospheres. The engine is expected to be capable of operation at chamber pressure of several atmospheres with a specific impulse level of 1100 seconds at .5 lbs. of thrust. Higher specific impulse operation can be obtained at the expense of reduced thrust capabilities.

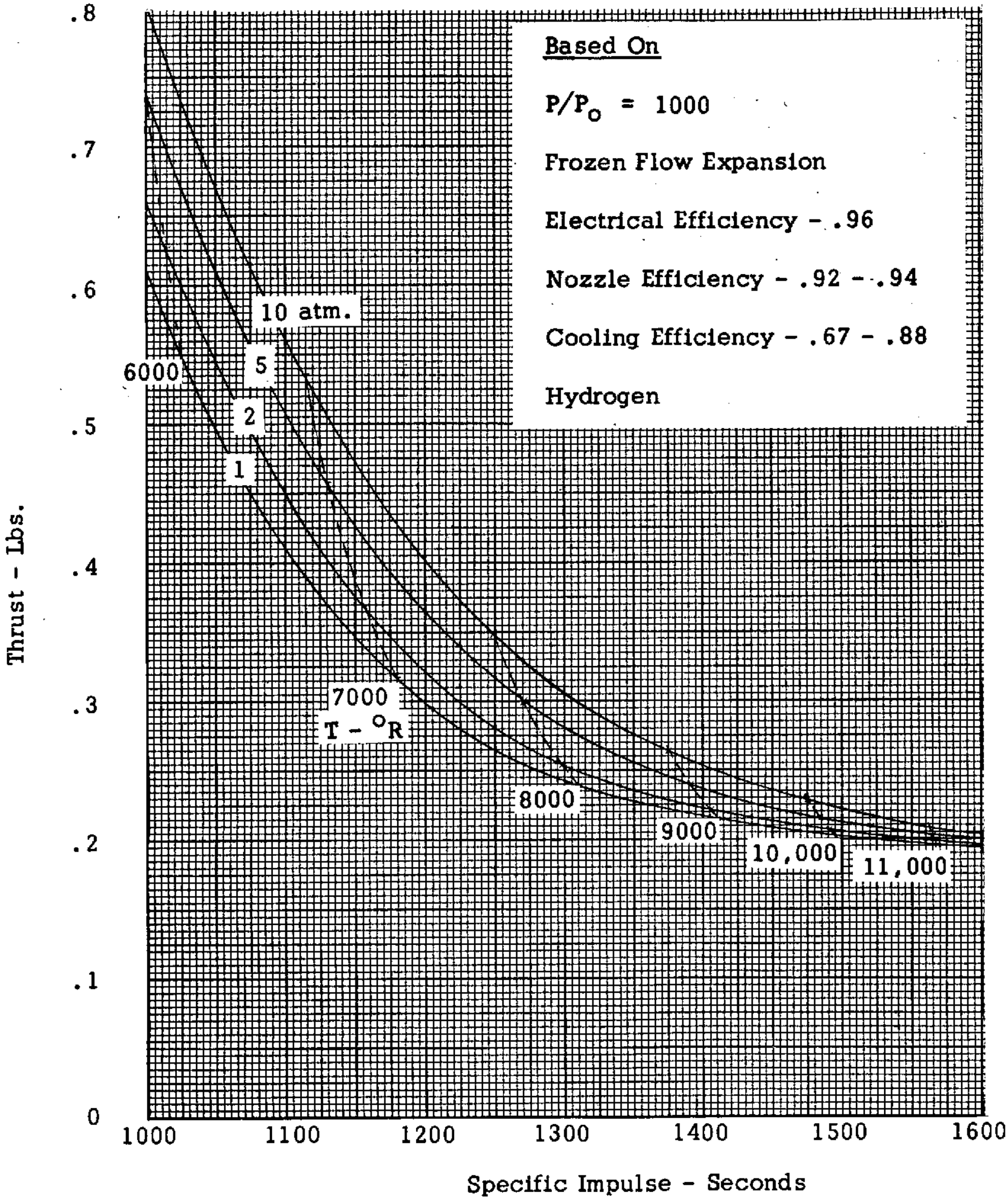

Figure 5.1 Expected 30 KW Hydrogen Arc Jet Engine Performance

A similar thermodynamic analysis was conducted for the ammonia cycle.[5] The higher chamber temperatures required for operation with ammonia and its poorer heat transfer properties require the use of a radiation cooled nozzle in place of the regeneratively cooled hydrogen nozzle. The higher heat losses associated with radiation cooling reduce the estimated nozzle efficiency level by about 10%. The following component efficiencies were, therefore, utilized in conjunction with the ideal performance obtained:

Electrical System Efficiency	- 96%
Nozzle Efficiency	- 92-94%
Radiation Cooling System Efficiency	- 70-50%

The results achieved are illustrated in Figure 5.2 for a range of specific impulse from 400 to 1150 seconds. Note that the maximum specific impulse which is expected to be attainable at .5 lbs. of thrust is 730 and that the maximum thrust expected at 1000 seconds specific impulse is .4 lbs. Thus ammonia engine operation is expected to result in performance capabilities below the current engine specifications.

5.2 Experimental Performance

Experimental XT-761 engine performance obtained with both hydrogen and ammonia propellants are indicated in Figure 5.3. The predicted performance shown in Figures 5.1 and 5.2 have been superimposed upon these data in order to permit comparisons.

The experimental hydrogen engine performance appears to be somewhat better than that predicted. This can be attributed to the attainment of somewhat better cooling efficiency than that predicted and to the possibility of regaining some of the energy assumed to be tied up in the gas dissociation due to recombination during the expansion process.

The single experimental ammonia test point obtained appears to be somewhat poorer than the levels predicted. This discrepancy may be due to the use of a regeneratively cooled nozzle configuration for running the test in comparison with performance predictions based on the assumption of a radiation cooled nozzle. Additional tests would be necessary, however, before any valid conclusions can be reached.

In general, however, the experimental results achieved would appear to substantiate the level of component efficiencies selected for generating the predicted performance.

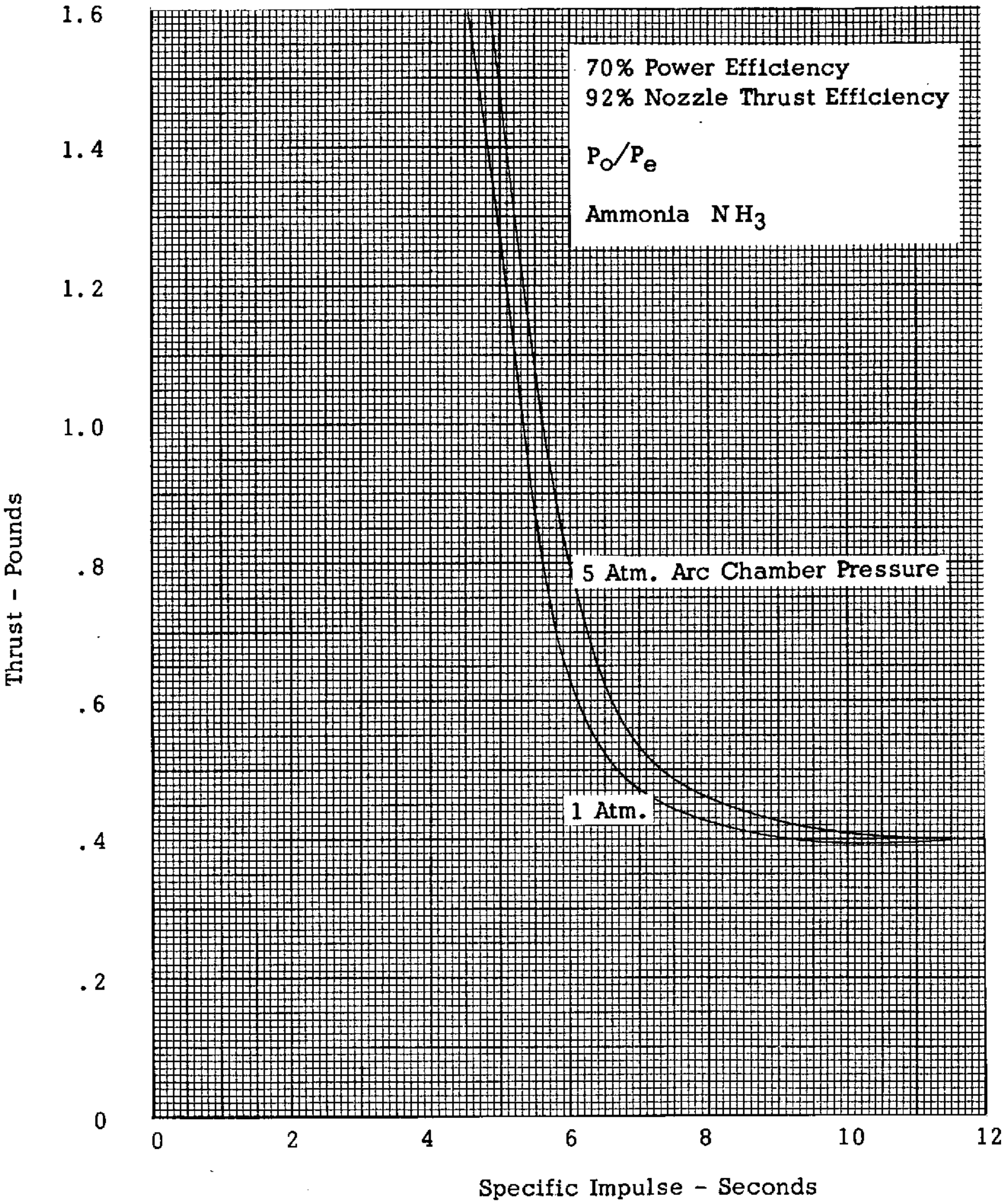

Figure 5. 2 Expected 30 KW Ammonia Arc Jet Engine Performance

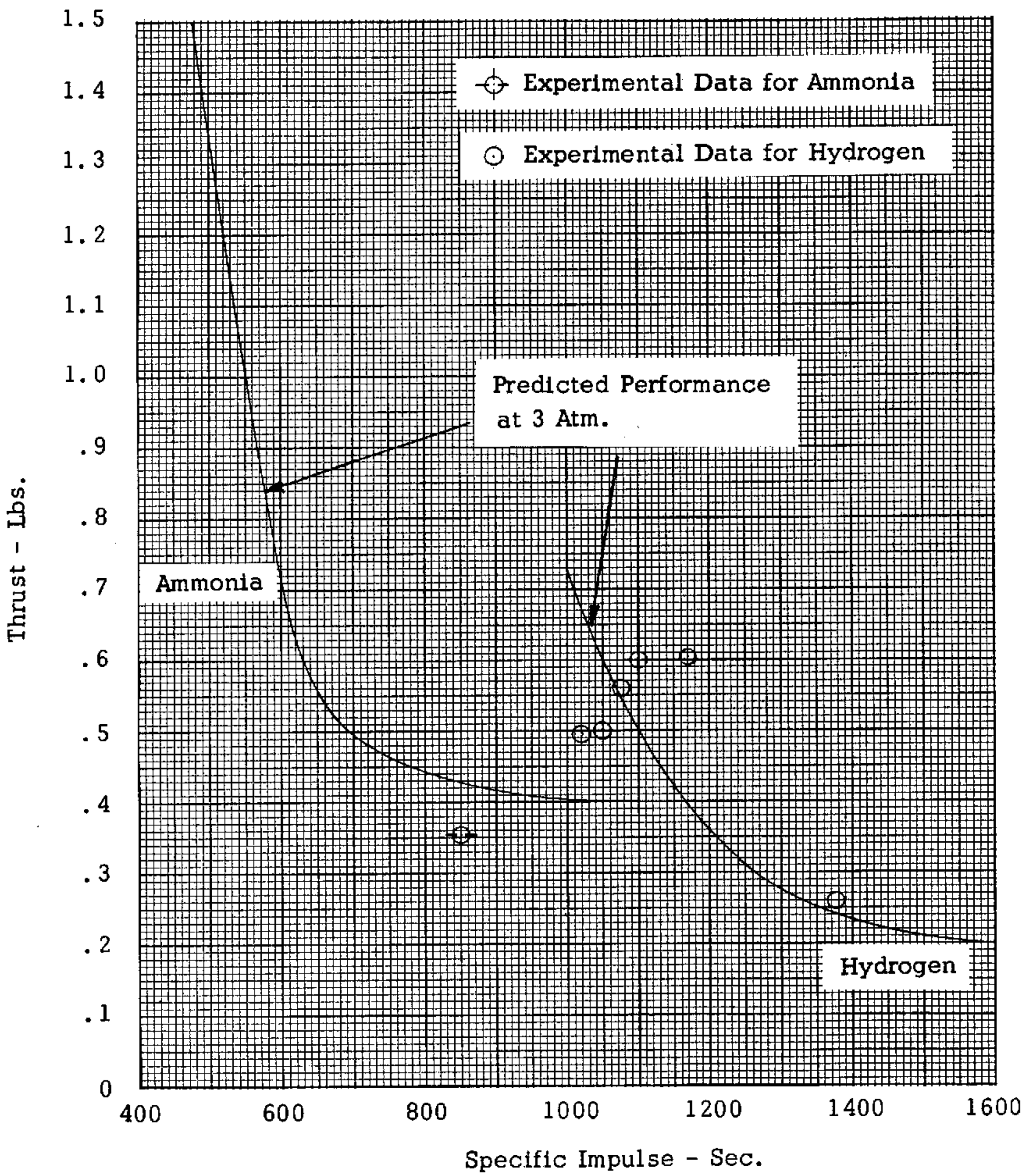

Figure 5.3 Experimental Engine Performance - 30 KW Arc Jet

6. MISSION ANALYSIS

Generic candidate missions have been defined and studied in order to identify propulsion system requirements and mission performance capabilities. The candidate missions selected include:

1. Stationary Satellite Transfer,
2. Lunar Satellite Transfer, and
3. Satellite Network Distribution.

These candidate missions represent a range of propulsion requirements which are typical of specific NASA and DOD space programs the arc jet propulsion system is capable of performing.

Each of the candidate missions were analyzed in detail. The individual propulsion tasks required for each mission were identified. Vehicle configuration requirements were determined and the results of the component and systems studies utilized to define equipment weight requirements.

Trajectory calculations were then obtained with the aid of the Space Trajectory Program (6) developed by the Flight Propulsion Laboratory Department. This program is an IBM-7090 digital computer procedure designed specifically for precision calculations of electrical propulsion system trajectories. The trajectory characteristics obtained were then utilized to generate parametric mission performance data for a wide range of mission requirements and arc-jet engine performance characteristics. The specific performance range predicted for the XT-761 arc-jet engine has been either superimposed upon these curves or indicated on separate curves.

The parametric mission performance data are based upon the use of hydrogen as the engine propellant. Additional mission performance data are included in order to compare hydrogen capabilities with those of ammonia. The results of these comparisons are utilized to recommend propellant selection.

6.1 Stationary Satellite Transfer

This mission involves the transfer of a satellite from an initial 300 nautical mile circular orbit to a final 22,240 mile altitude circular orbit - the 24 hour orbit. An Atlas-Centaur with an assumed orbital payload capability of 8500 lbs. was utilized for the launch vehicle. An alternate mission studied included a 28° inclination correction along with the above altitude transfer. The final orbit in this case would be a stationary satellite orbit. This mission would be applicable to the Advent, Relay II, Commercial Communication Satellite, Aeros, and Astrostat programs currently under development or in the planning stage.

6.1.1 Vehicle Configuration

Both 30 and 60 KW Snap 8 configurations have been studied. The 60 KW configuration is used to operate two 30 KW arc jet engines simultaneously. The engines are located at the aft end of the vehicle and at opposite ends of its diameter. Thrust vectoring can be utilized to provide attitude stabilization about pitch, yaw, and roll axes.

A single arc jet engine is used with the 30 KW Snap 8 configuration. It is positioned at the aft end of the vehicle and on its axis. The single engine is capable of providing attitude stabilization about only the pitch and yaw axes. Additional engines - of 1 KW power rating - must be utilized to provide roll control. A system of four roll engines will eliminate any necessity for vectoring or turning the engines.

A summary of component and system weights is indicated in Table 6.1.

6.1.2 Mission Profile

A typical satellite transfer mission profile is illustrated in Figure 6.1. These results show a transfer from an initial 300 nautical mile parking orbit to a final 22,240 mile altitude orbit using a 60 KW engine configuration. The engine performance is based upon 1.0 lb. of thrust and a specific impulse of 1250 seconds.

Transverse thrust is utilized to generate the required transfer orbit during the first 906 hours of flight. [7] The engine is then shut down and the vehicle allowed to coast to the 24 hour orbit altitude during the next 6 hours. A final 2.5 hour propulsion period is utilized to obtain a circular orbit.

The absence of re-start capability of the Snap 8 power system will require a slight deviation from this optimum flight plan in order to permit continuous propulsion. This will not, however, significantly effect the results obtained.

TABLE 6.1

COMPONENT AND SYSTEM WEIGHTS

Snap 8 Power Level	30 KW	30 KW	60 KW	60 KW
No. of 30 KW Arc Jet Engines	1	2	2	4
Mission	Satellite Transfer	Satellite Distribution & Lunar Transfer	Satellite Transfer	Lunar Transfer
Power Supply - Lb.	2000	2000	3000	3000
Propulsion Engines - Lb.	4	8	8	16
Transformer - Lb.	90	90	180	180
Inductances - Lb.	6	6	12	12
Starter - Lb.	8	8	12	12
Switches - Lb.	4	8	8	16
Flow Control - Lb.	2	3	4	6
Thrust Vectoring - Lb.	8	15	15	30
Roll Control - Lb.	5	5	--	--
Total	2127 Lb.	2143 Lb.	3239 Lb.	3272 Lb.

Time - Hours	Thrust Pattern
0 - 906	Transverse
906 - 912	Coast to Apogee
912 - 914.5	Circularize

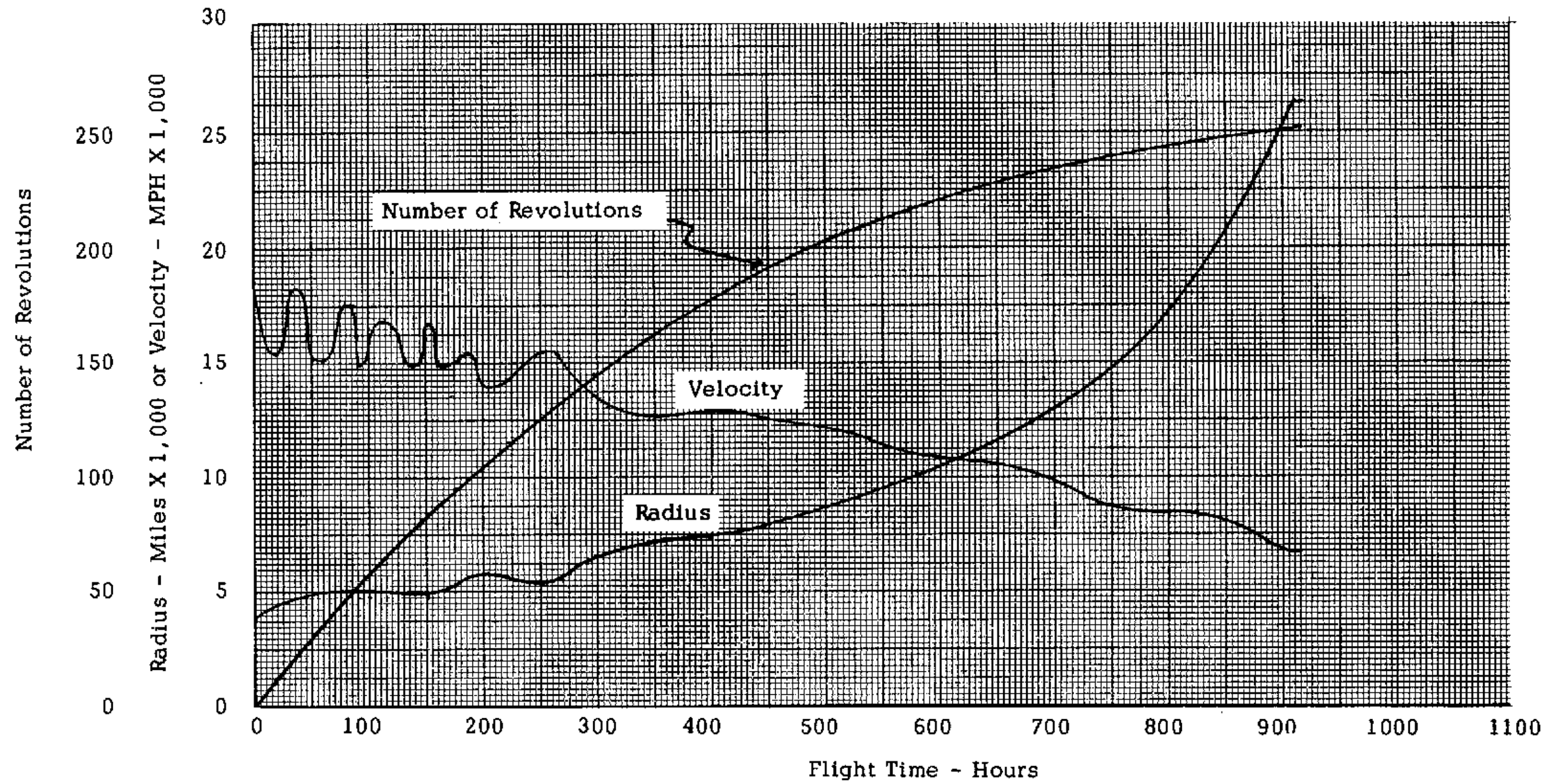

Figure 6.1 24 Hour Satellite Transfer Mission.

Trajectory characteristics for satellite transfer to a stationary satellite orbit - 24 hour equatorial orbit - are illustrated in Figure 6.2. Thrust is applied at a 40° angle with respect to the instantaneous orbit plane in order to achieve a 28° inclination correction during the initial propulsion period to generate the required transfer orbit. This results in an increase in the duration of the initial propulsion period to 1120 hours. The coasting and final propulsion period requirements remain 6 and 2.5 hours respectively.

6.1.3 Performance

Additional trajectory calculations were obtained for other combinations of engine thrust and specific impulse. These results were correlated and utilized to develop simplified expressions for the overall mission propulsion time requirements. The procedure and the resulting expressions are described, in more detail, in the Appendix.

These expressions were then utilized to develop parametric mission performance data. The results are indicated in Figure 6.3. A specific impulse range of 1000 to 1500 seconds and a final altitude range from 2000 to 22,240 miles are illustrated. Any desired thrust can be obtained from the total impulse lines - the product of thrust and propulsion time - which are superimposed on the curve. It should be noted that the thrust level will effect only the propulsion time required to complete a given mission and not the performance capability of the mission as measured by the gross payload obtainable. Initial altitude has been held constant at 300 nautical miles.

The payload indicated includes the weight of the Snap 8 power system but not the weight of either the engine or the equipment required for engine operation. If the Snap 8 power system cannot be utilized by the remaining payload at either its full power level or at some reduced power level, 2000 lbs. should be subtracted from the gross payload for a 30 KW configuration and 3000 lbs. for a 60 KW configuration to obtain the net effective payload.

Similar satellite transfer payload capabilities for combined altitude transfer and inclination correction are illustrated in Figure 6.4. The total inclination correction has been held constant at 28°. Thus, all final orbits will be equatorial orbits if launched eastward from Cape Canaveral.

A nominal orbital weight capability of 8500 lbs. for a 300 nautical mile orbit has been utilized for the Atlas-Centaur boost vehicle. This level is expected to be achieved with a payload package capable of being stored within the 11 foot nose cone shown in Figure 4.1. The expected 28 to 35 foot length of the arc-jet vehicle required for containing the Snap 8 and the propellant storage tank in addition to the payload may result in a reduced orbital weight capability. It is recommended that this be explored in more detail with the boost vehicle manufacturer.

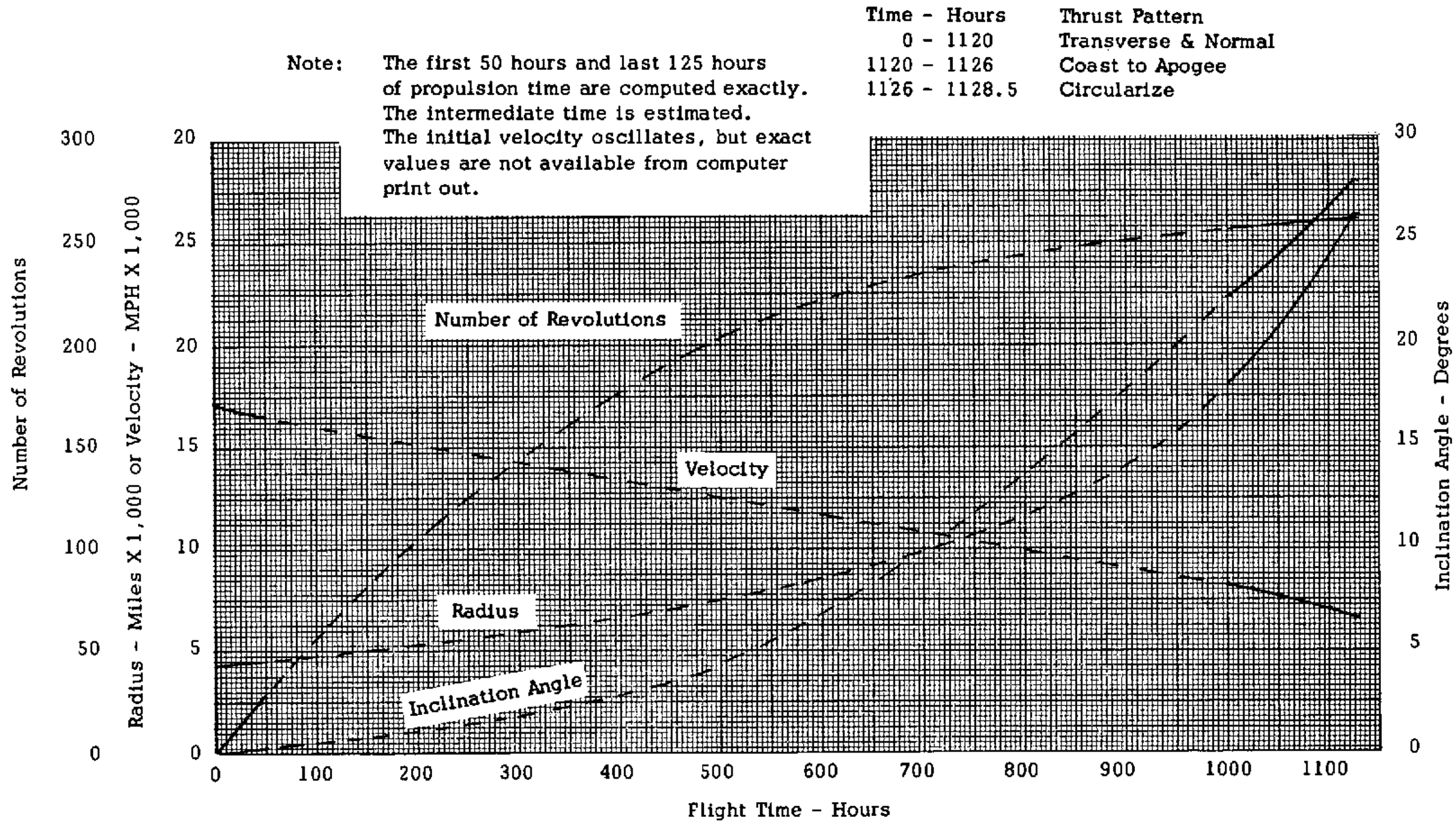

Figure 6.2 Stationary Satellite Transfer Mission.

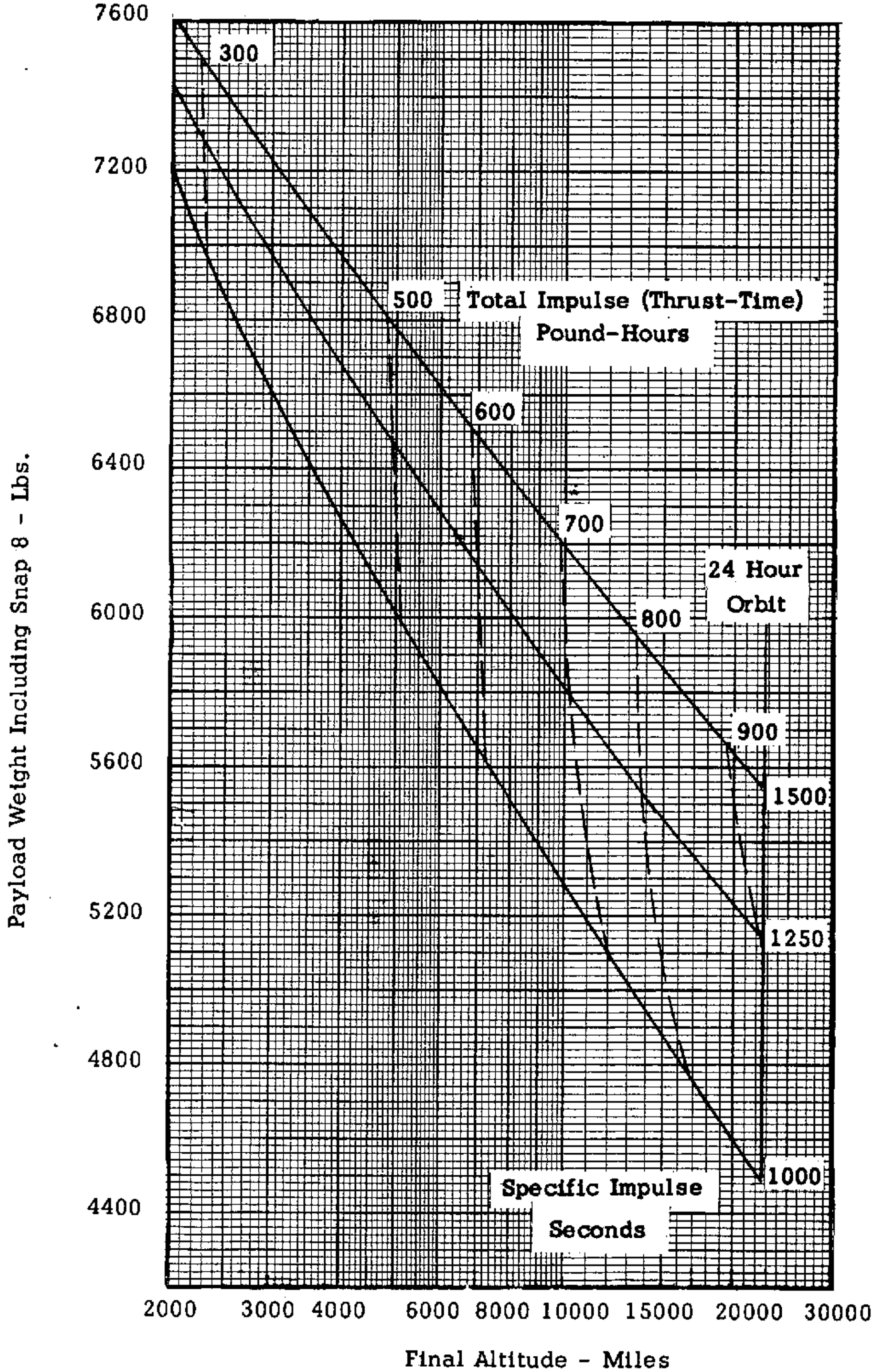

Figure 6.3 Payload Capabilities for Satellite Transfer Mission. No Inclination Correction. Centaur Boost Vehicle.

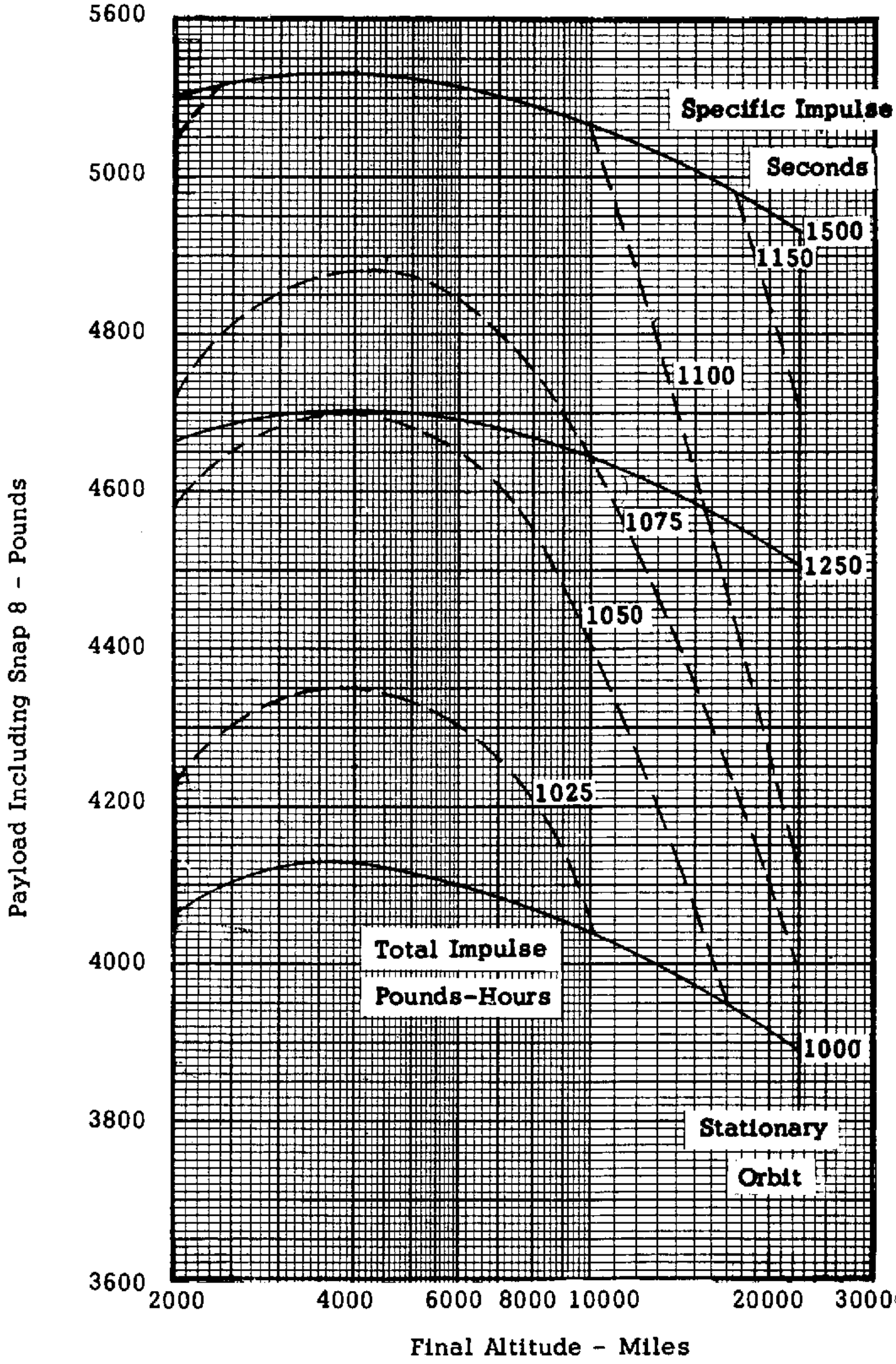

Figure 6.4 Payload Capabilities for Satellite Transfer Mission. 28° Inclination Correction. Centaur Boost Vehicle.

The effects of reduced launch vehicle orbital payloads on specific satellite transfer mission capabilities are illustrated in Figures 6.5 and 6.6. Figure 6.5 presents the effects of initial orbital weight on the gross payload (scientific payload plus power supply weight) transferred to a stationary satellite orbit. Figure 6.6 presents similar data for the transfer to a 24 hour orbit. Both reflect the performance of the basic vehicle configuration.

It is noted that for the stationary satellite transfer, the payload is reduced approximately 500 pounds for each 1000 pound reduction in the Centaur orbital boost capability.

For the 24 hour satellite transfer mission the results indicate a reduction of 600 pounds of payload for each 1000 pound reduction in initial orbital vehicle weight.

For the shared tank vehicle configuration, the payload deterioration which is due to reduced initial orbital weight is somewhat greater than that indicated for the basic vehicle design. Specifically, for the stationary satellite transfer and the 24 hour satellite transfer missions the payload decrements per 1000 pounds of initial vehicle weight are approximately 560 and 650 pounds respectively.

6.2 Lunar Satellite Transfer

This mission involves the transfer of a satellite from an initial 300 nautical mile orbit about the earth to a 20 mile altitude circular orbit about the moon. An Atlas-Centaur with an assumed orbital payload capability of 8500 lbs. was utilized for the launch vehicle. This mission would be applicable to the current Surveyor program under development and to a dust-return version of the Surveyor currently in the planning stage.

6.2.1 Vehicle Configuration

The 30 and 60 KW satellite transfer configurations could also be used for the lunar satellite transfer mission. Thrust orientation requirements during the lunar capture phase of the mission require a thrust direction reversal once during each revolution about the moon. In order to eliminate added control complexities imposed by these requirements, additional 30 KW engines have been utilized. Thus, the 30 KW configuration would utilize one engine at each end of the vehicle and the 60 KW configuration would use two engines at each end. Power would be switched from one engine set to the other as required. The 30 KW configuration will require 4-1 KW roll engines for roll control as before.

Component and system weights are summarized in Table 6.1.

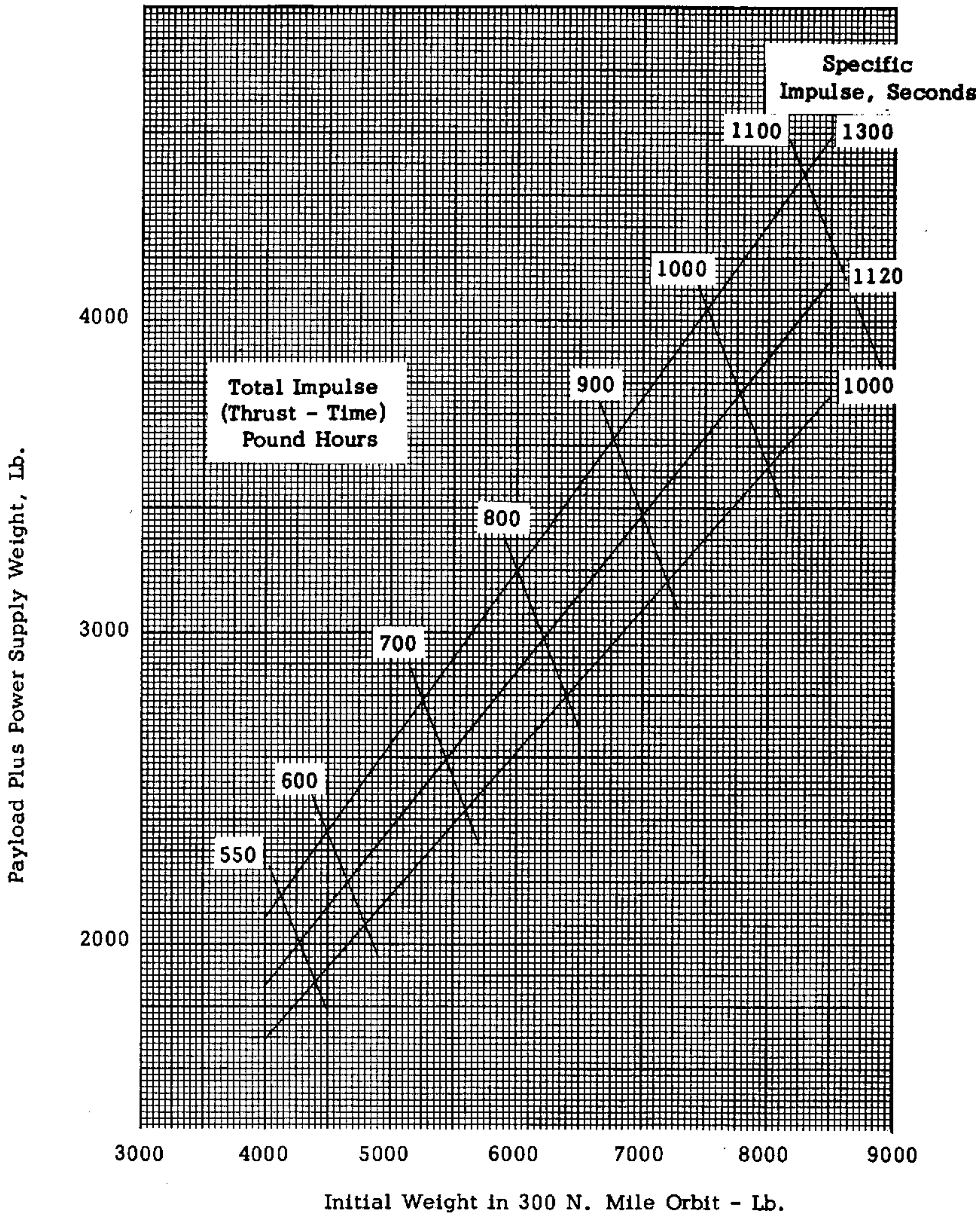

Figure 6.5 Effect of Centaur Low Orbit Payload Capability on Gross Payload Transferred by 30 KW Arc Jet to a Stationary Satellite Orbit.

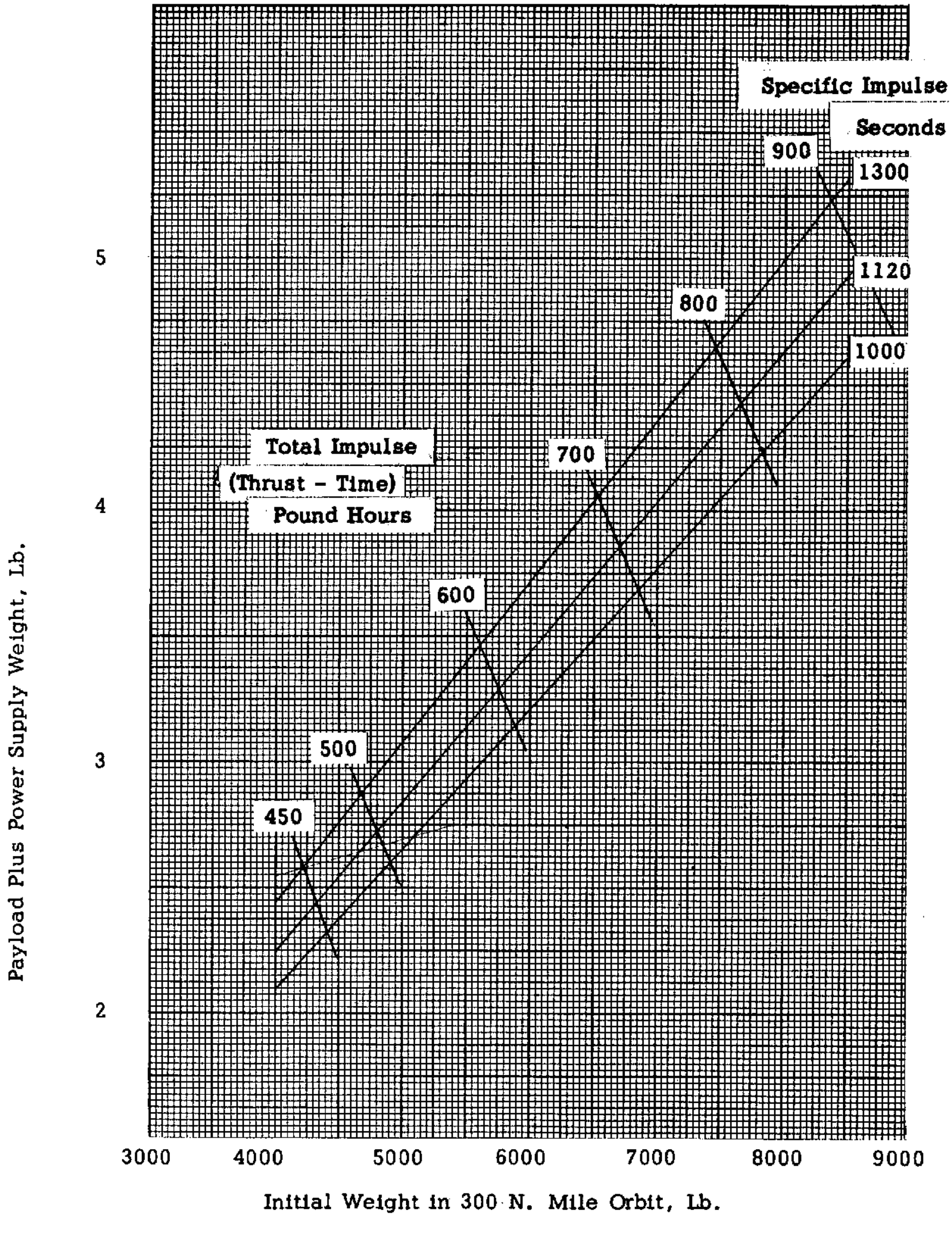

Figure 6.6 Effect of Centaur Low Orbit Payload Capability on Gross Payload Transferred by 30 KW Arc Jet to a 24 Hour Satellite Orbit.

6.2.2 Mission Profile

A typical lunar satellite transfer mission profile is illustrated in Figures 6.7 and 6.8. Figure 6.7 contains the earth escape and coasting phases of the lunar trajectory. The propulsion period is a continuation of the transverse propulsion period utilized for the 24 hour transfer. Figure 6.8 illustrates the lunar capture trajectory during the final propulsion period. The thrust orientation program required to minimize propulsion requirements for achieving a 20 mile altitude orbit includes the use of a constant pericenter thrust orientation during the major part of the capture maneuver.[7] This pattern requires a continuously variable thrust orientation with a thrust direction reversal at the completion of each revolution.

Substantial lunar perturbations were detected during the latter part of the coasting period and similar earth perturbations during the lunar capture phase. The magnitude of these perturbations suggests caution in the use of two-body trajectories for anything but initial feasibility studies.

The trajectory shown is based upon an engine thrust of 1.0 lbs. a specific impulse of 1250 seconds, and a 60 KW engine configuration. The trajectory required an initial propulsion period of 1191 hours, an intermediate coasting period of 76 hours, and a final propulsion period of 242 hours. This will either necessitate the addition of re-start capability to the Snap 8 power system or else the use of a dummy load to provide for power dissipation during the coasting period.

6.2.3 Performance

Parametric mission performance capabilities for transfer to a 20 mile orbit about the moon are illustrated in Figure 6.9. Payload capabilities indicated include the weight of the Snap 8 power system but not the weight of engines or other equipment required for operation of the engines. Performance is included for both 30 KW and 60 KW engine configurations, for a thrust range of .5 to .75 lbs. per 30 KW, and for a specific impulse range of 1000 to 1500 seconds. The performance range predicted for the XT-761 engine is illustrated in the cross-hatched areas.

Note that the 60 KW Snap 8 weight of 3000 lbs. is greater than the 2880 lb. payload capability of the XT-761 engine. Consequently, the mission could not be completed with the 60 KW configuration. The 2000 lb. weight of the 30 KW configuration would have a net payload capability of 880 lbs. This payload capability would appear to be extremely marginal.

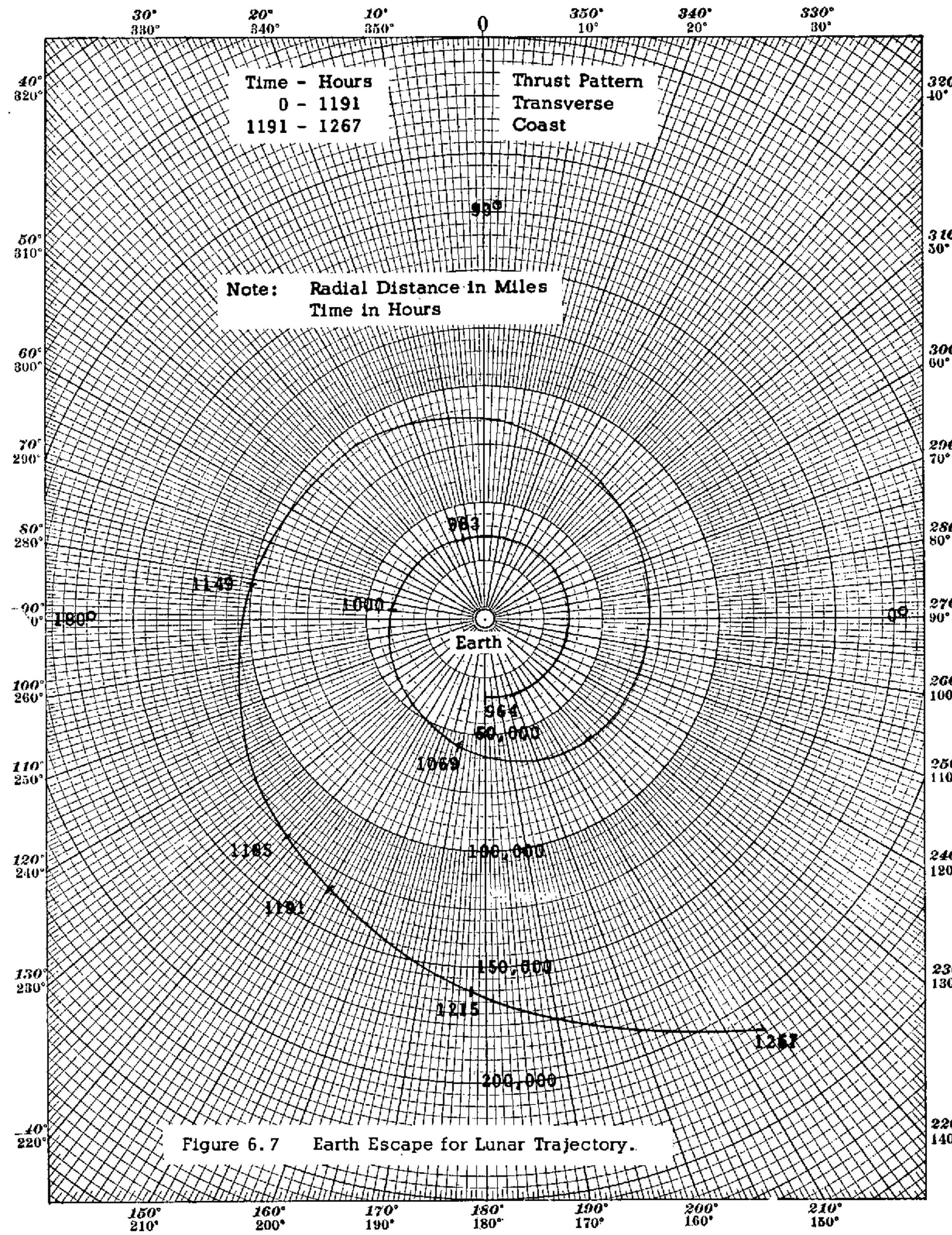

Figure 6.7 Earth Escape for Lunar Trajectory.

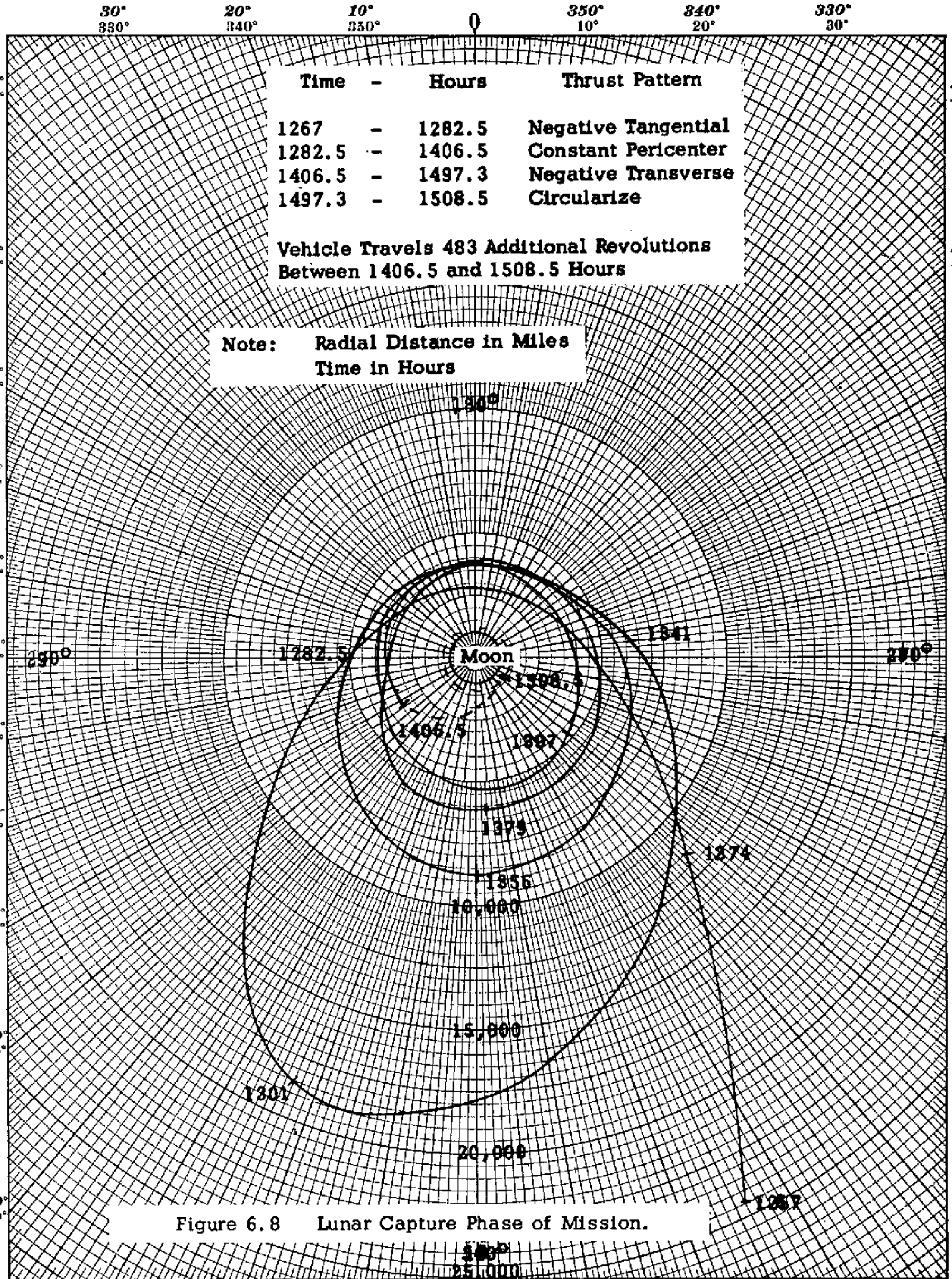

Figure 6.8 Lunar Capture Phase of Mission.

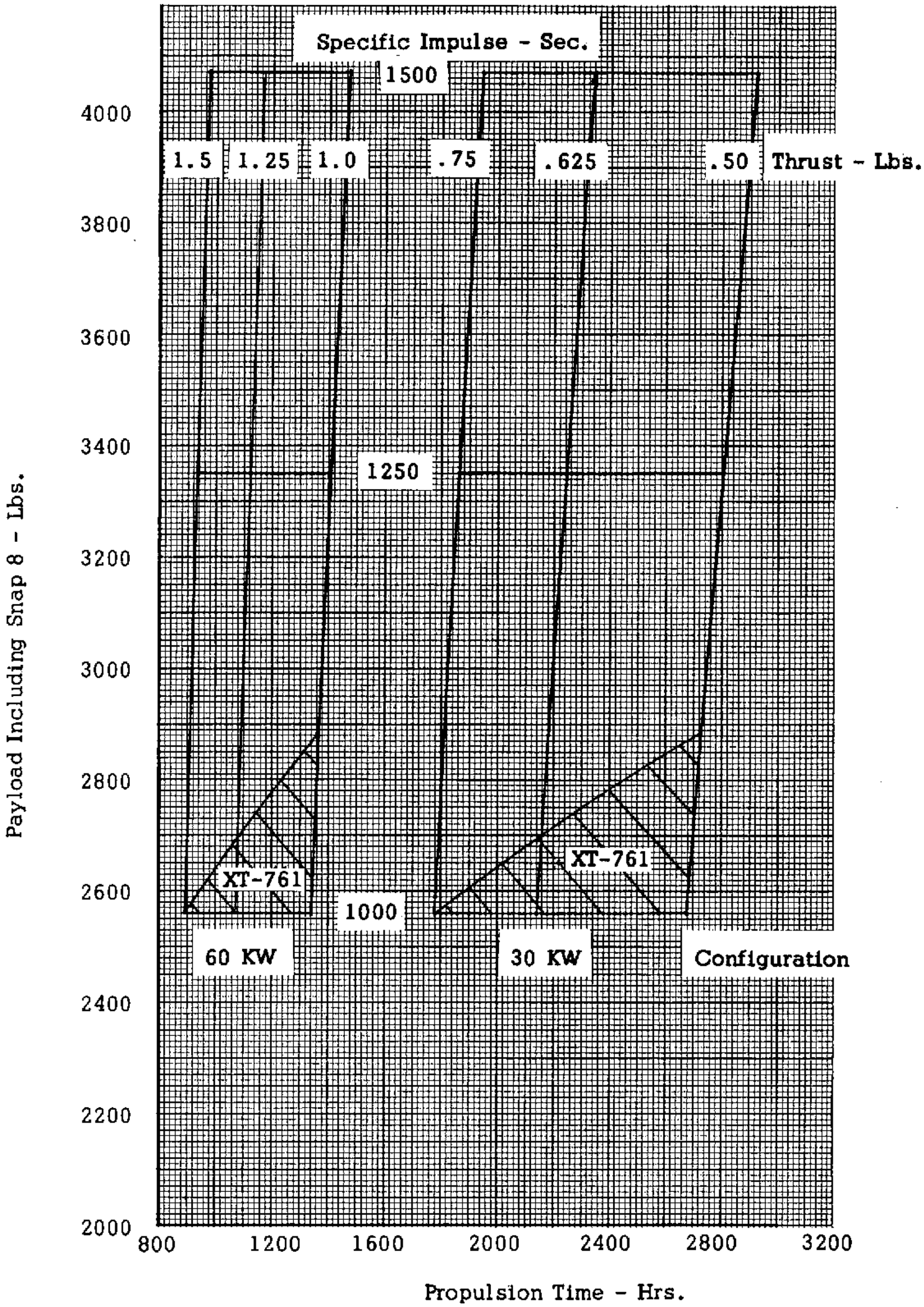

Figure 6.9 **Payload Capabilities for Lunar Transfer Mission. Centaur Booster.**

6.3 Satellite Network Distribution

This mission utilizes a single carrier vehicle to distribute a multi-satellite network at equally spaced intervals about a common orbit thus providing a multiple launch capability. A range of distribution altitudes from 300 to 16,000 miles has been investigated.

Propulsion requirements can be minimized by using the launch vehicle to inject the carrier vehicle into orbit at the desired distribution altitude. Substantially greater payload capabilities can be achieved by using the launch vehicle to inject the carrier vehicle into a 300 mile parking orbit. The arc jet propulsion system is then utilized to complete the transfer to the distribution altitude and for the distribution operation. The capabilities of the Saturn C-1 launch vehicle were investigated for both mission profiles and the capabilities of the Centaur for the latter profile. Orbital payload capabilities of 8500 lbs. and 19,000 lbs. at the 300 mile orbit were utilized for the Centaur and Saturn C-1 respectively.

These missions would be applicable to the Commercial Communication Satellite, Rebound, Midas, Samos, and Bambi programs currently under development or in the planning stage.

6.3.1 Vehicle Configuration

Although propulsion requirements for the stationary and lunar satellite transfers are relatively insensitive to variations in engine thrust level, requirements for satellite distribution increase as thrust level is increased. There is, therefore, no incentive to use the heavier 60 KW Snap 8 power system for this mission. Consequently, only a 30 KW configuration has been studied.

The distribution operation requires periodic thrust reversals - two for each satellite distributed. The two engine 30 KW configuration described in the previous section was, therefore, utilized.

6.3.2 Mission Profile

Trajectory characteristics are shown schematically for a typical satellite distribution mission in Figure 6.10. This trajectory is for the distribution of a network of 10 satellites about a 6000 mile altitude orbit. Calculations were based upon an initial carrier vehicle weight of 6550 lbs. - compatible with the payload capabilities of the Saturn C-1 launch vehicle for a 6000 mile satellite. An engine thrust of .5 lbs. and a specific impulse of 1250 seconds were utilized.

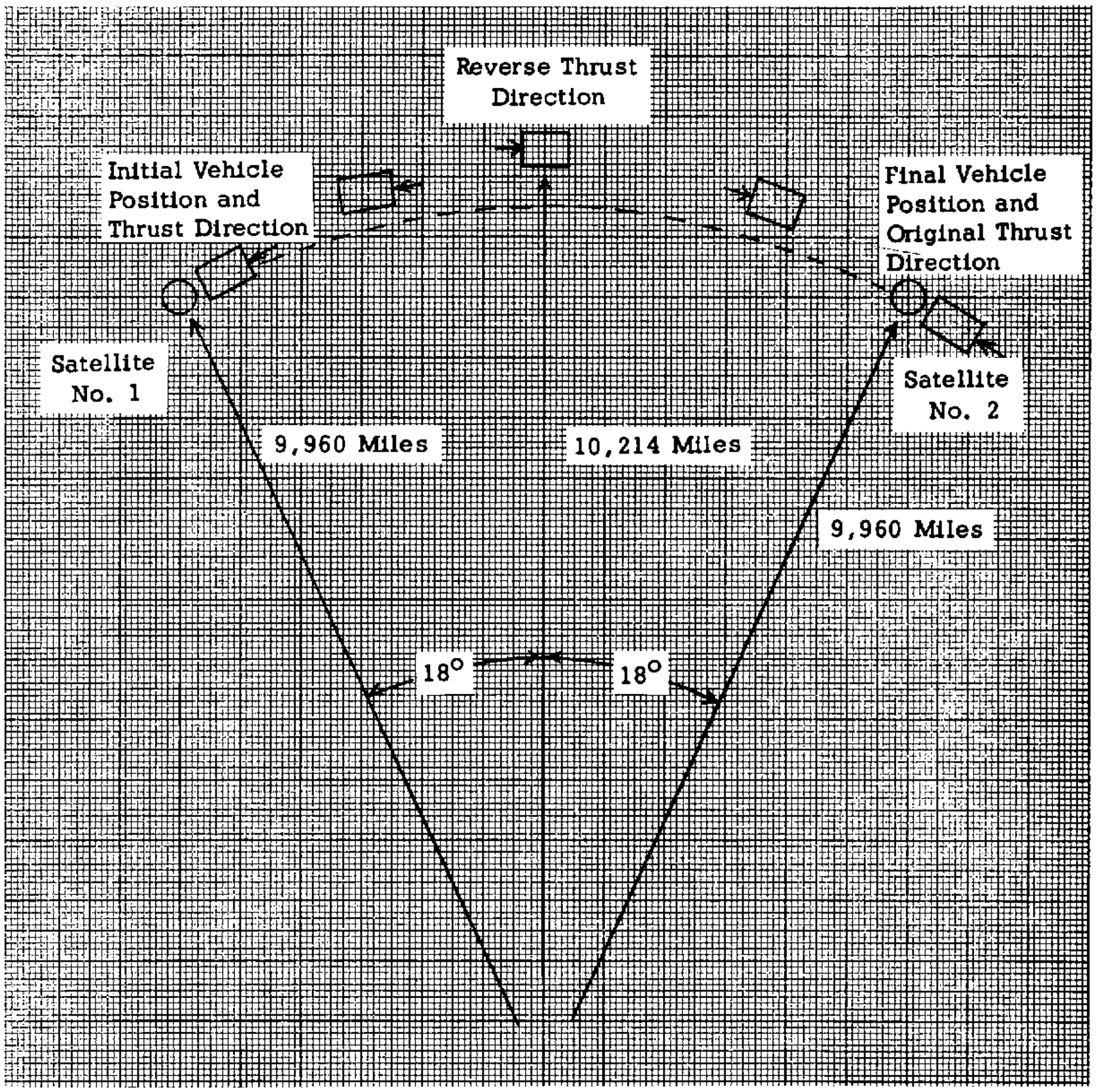

Figure 6.10 Satellite Distribution Mission, 10 Satellites in 6000 Mile Altitude Orbit.

The mission involves the release of the first satellite prior to starting the engine. An initial transverse thrust propulsion period of 18.2 hours is then used to displace the carrier vehicle from the first satellite by a central angle of 18°. [8] Thrust direction is then reversed 180° and applied for an additional 18.2 hours thereby displacing the carrier vehicle an additional 18° and returning it to its initial orbit altitude. The second satellite is then released and the process repeated - with gradually diminishing propulsion periods - until all ten satellites have been released.

These calculations indicate that 10 satellites, each weighing 375 lbs., can be distributed in this fashion at 6000 miles altitude. Propulsion time requirements were only 275 hours.

6.3.3 Performance

Satellite distribution mission performance capabilities are presented in Figure 6.11 as a function of distribution altitude for a 30 KW propulsion system operating at 0.5 lb. thrust and 1250 sec. specific impulse. These data are nominally for the distribution of a 10 satellite network. The Saturn C-1 line involves chemical boost to the desired distribution altitude and arc jet propulsion only for the distribution operation. The Centaur-arc jet and Saturn C-1 arc jet lines represent chemical boost to a 300 mile parking orbit with arc jet propulsion for transfer to the distribution altitude and for the distribution operation as well. Note that the weight distribution capabilities of the Centaur-arc jet exceed those of the Saturn C-1 for distribution altitudes above 5700 miles. Since the Snap 8 power supply remains in the carrier vehicle, it is not included in the Satellite weights indicated.

Propulsion time requirements associated with each of the above configurations are illustrated in Figure 6.12 as a function of distribution altitude. Note that attractive distribution missions can be completed in propulsion times as low as 245 hours (about 10 days).

Second order effects on weight distribution capabilities are presented in Figure 6.13. The top curve contains correction factors for satellite networks of 10 to 40 satellites. The middle curve contains correction factors for thrust levels other than 0.5 lb. Note that satellite weight capabilities are reduced as thrust is increased. The bottom curve illustrates the effect of specific impulse levels other than 1250 seconds. The corrections obtained should be added to the nominal data of Figure 6.11.

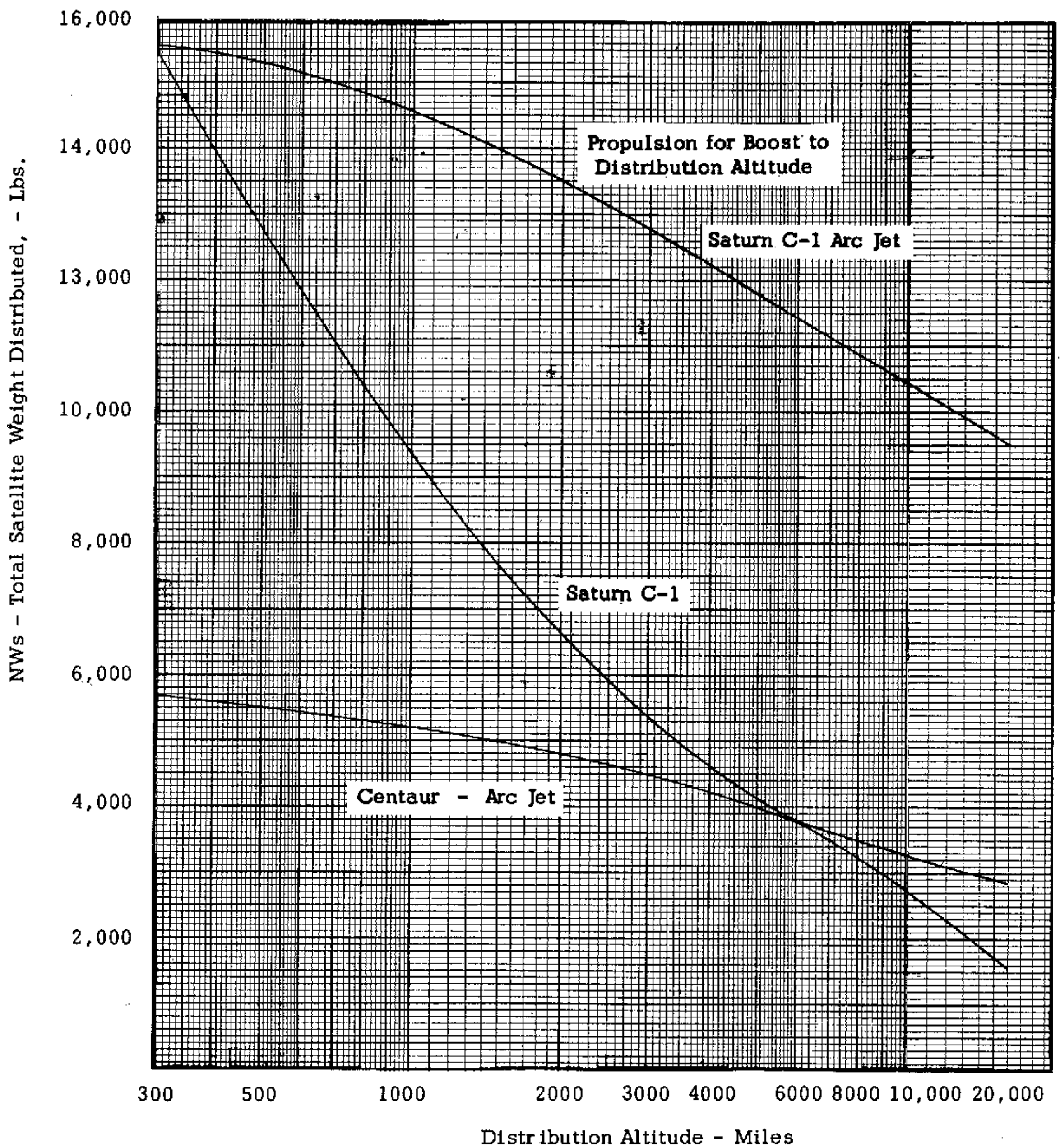

Figure 6.11 **Payload Capabilities for Satellite Distribution Mission. 30 KW Configuration - Thrust, .5 lbs., Specific Impulse.**

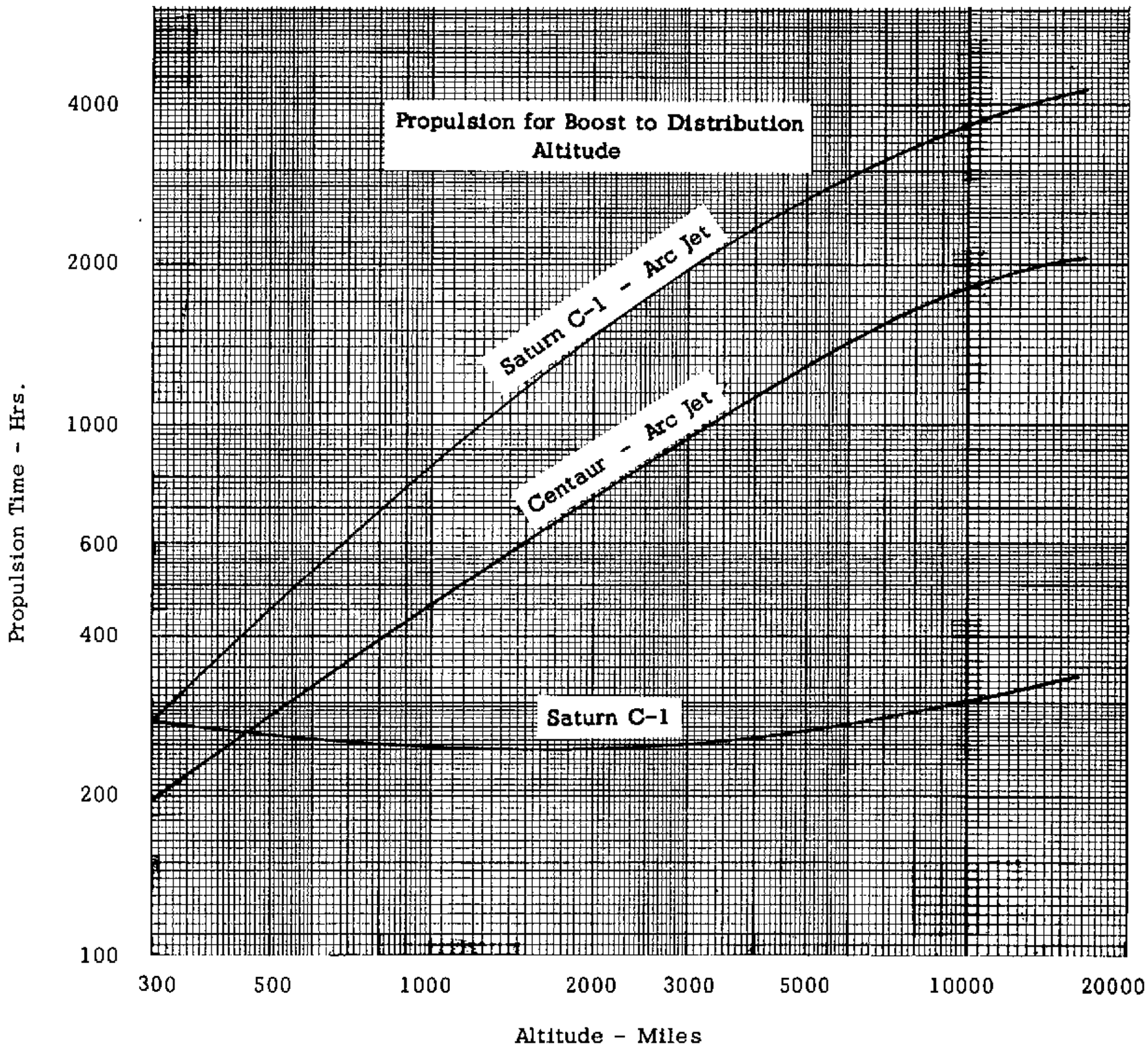

Figure 6.12 Propulsion Time Requirements for Satellite Distribution Mission. 30 KW Configuration. Thrust, .5 lbs. Specific Impulse, 1250 sec. 10 Satellite Network.

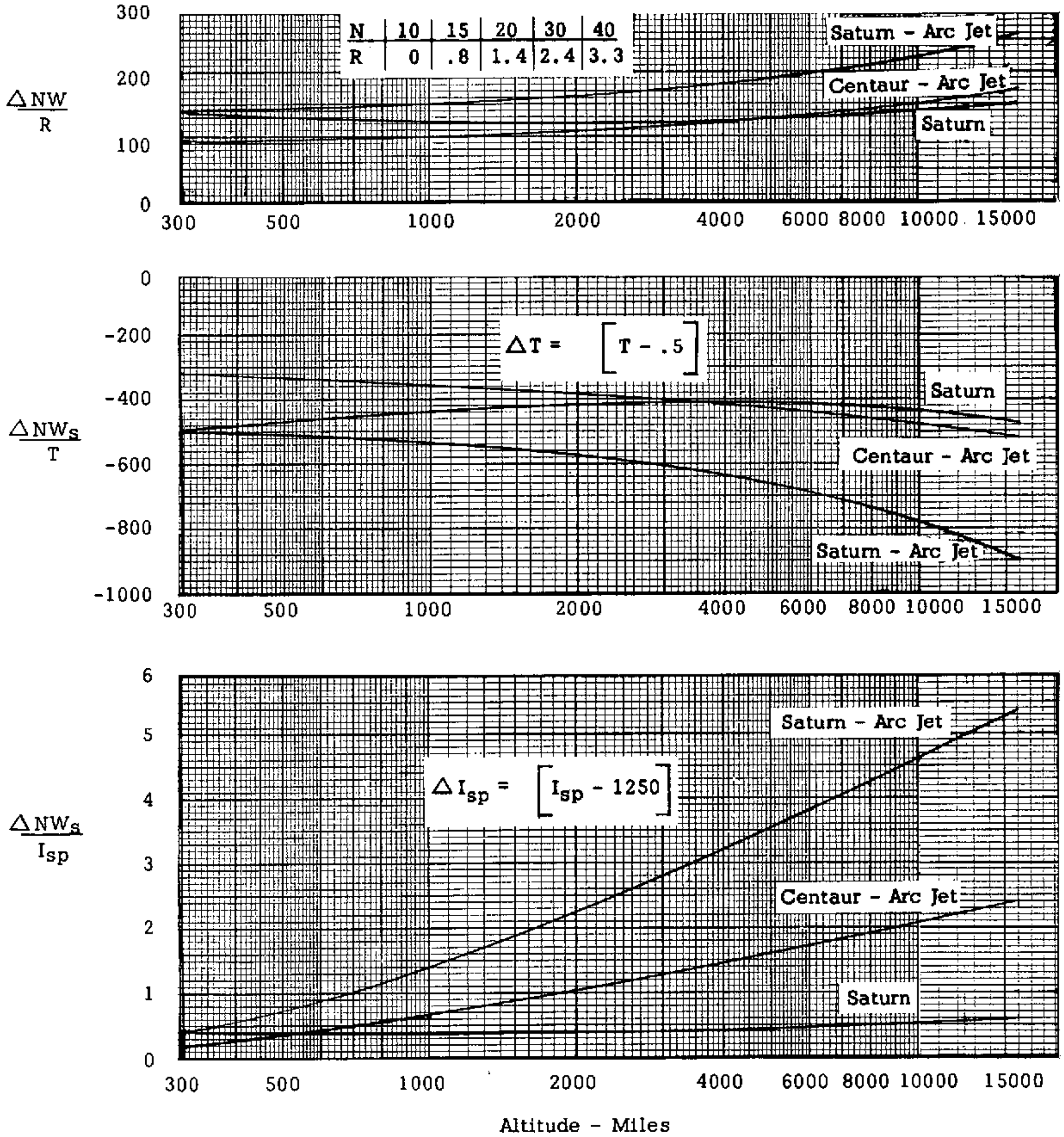

Figure 6.13 Satellite Weight Correction Factors for Satellite Distribution Mission.

The performance capabilities of the XT-761 engine are indicated in Table 6.2. The data indicated are for the distribution of a 10 satellite network at a final distribution altitude of 6000 miles. Both current engine performance and expected performance levels at the end of the engine development period are indicated.

6.4 Propellant Selection

Although the low molecular weight of hydrogen propellant provides extremely attractive engine performance characteristics, its other properties result in somewhat unfavorable system characteristics. These are due principally to its extremely low density and cryogenic storage requirements for storage in the liquid phase. Consequently, the properties and system performance capabilities of other suitable propellants were compared with those of hydrogen.

6.4.1 Alternate Propellants

The most attractive alternate propellant would appear to be ammonia. Its high hydrogen content results in an average exhaust molecular weight which is as low as any other possible propellant. Its storage properties are such that it can be stored under several atmospheres of pressure at ambient temperatures and its density is an order of magnitude greater than hydrogen.

Another possibility is helium. Its exhaust molecular weight is about the same as ammonia and its density in the liquid phase is some place between those of hydrogen and ammonia. Its thermal insulation requirements are somewhat more severe than those of hydrogen. Helium would, therefore, seem to combine all of the worst system features of hydrogen and ammonia.

Other possible propellants would result in higher molecular weights than those of ammonia with about the same or slightly better storage characteristics. It has been concluded, therefore, that a comparison of hydrogen and ammonia system performance capabilities would be sufficient to identify the most attractive propellant for the 30 KW arc jet propulsion system.

6.4.2 Stationary Satellite Transfer

The performance capabilities of hydrogen and ammonia propellants for the stationary satellite transfer mission are presented in Figure 6.14 as a function of specific impulse. The payloads indicated do not include the weight of the Snap 8 power system. Engine performance data have been obtained from Figures 5.1 and 5.2. It is evident that hydrogen operation is

TABLE 6.2

XT-761 PERFORMANCE CAPABILITIES FOR SATELLITE DISTRIBUTION

10 SATELLITES AT 6000 MILES ALTITUDE

	CURRENT PERFORMANCE	EXPECTED PERFORMANCE
Thrust - Lbs.	.5	.5
Specific Impulse - Sec.	1000	1100
Saturn C-1 Booster		
Weight per Satellite - Lbs.	365	369
Propulsion Time Requirements - Hrs.	275	275
Centaur - Arc Jet Booster		
Weight per Satellite - Lbs.	337	354
Propulsion Time Requirements - Hrs.	1410	1410
Saturn C-1 Arc Jet Booster		
Weight per Satellite - Lbs.	1054	1092
Propulsion Time Requirements - Hrs.	2950	2950

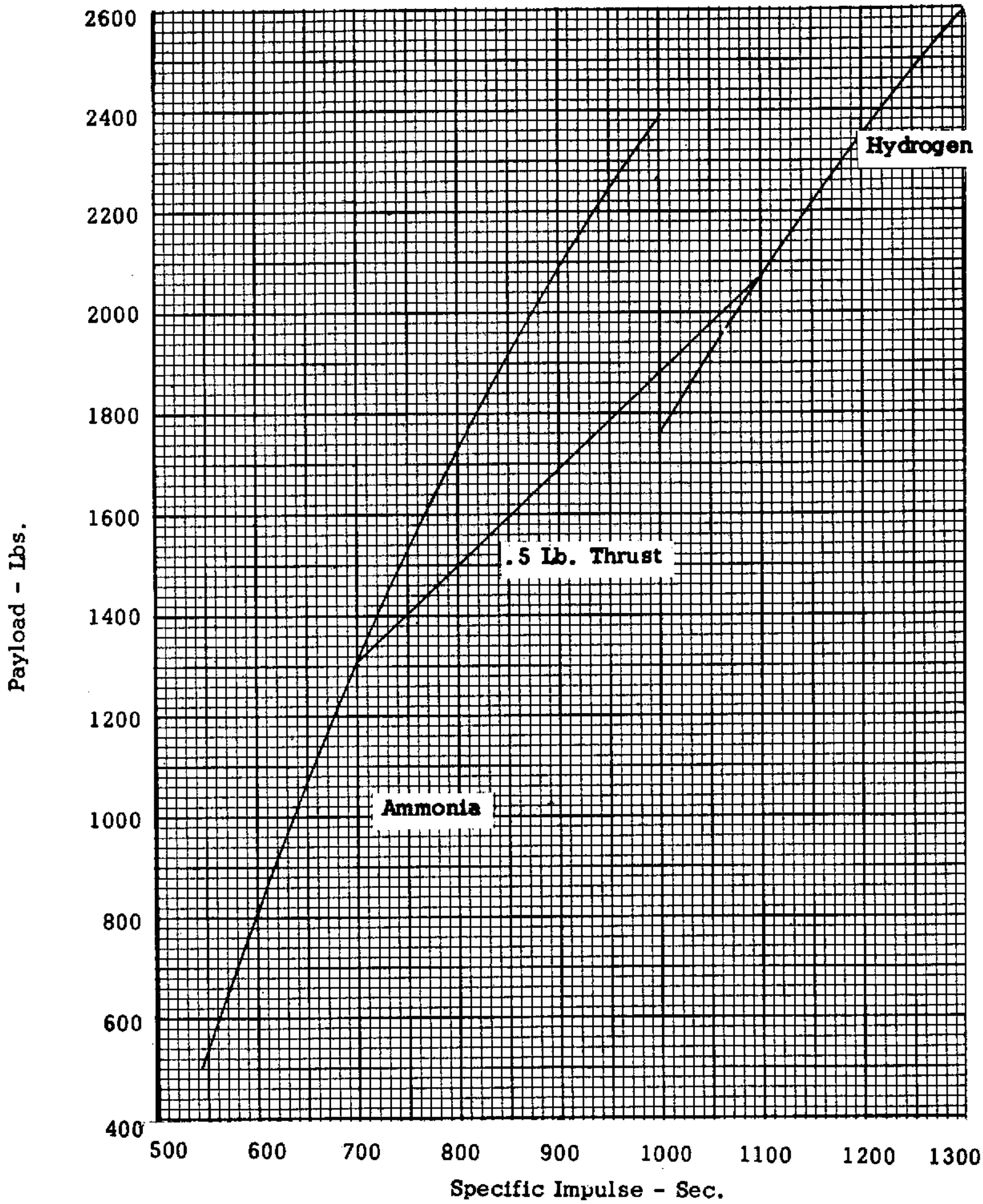

Figure 6.14 Stationary Satellite Transfer Mission Capabilities. Centaur Booster and 30 KW Arc Jet.

required to be at 200 seconds higher specific impulse in order to match performance capabilities with those of ammonia. Such increased specific impulse operation is, however, readily obtainable. The ultimate specific impulse level that can be obtained with hydrogen is expected to be approximately twice the level that can be obtained with ammonia.

The current XT-761 engine performance specifications call for a minimum thrust of .5 lbs. and a minimum specific impulse of 1000 seconds. These specifications cannot be achieved with ammonia. The maximum specific impulse at which .5 lbs. of thrust can be achieved is 700 seconds with ammonia and 1100 seconds with hydrogen. At these levels, hydrogen performance capabilities are 61% greater than those of ammonia.

6.4.3 Satellite Network Distribution

Similar comparison data are presented in Figure 6.15 for the satellite network distribution mission. These data indicate capabilities for transferring the required carrier vehicle from an initial 300 mile parking orbit to a 6000 mile altitude orbit and then distributing a 10 satellite network. The Saturn C-1 launch vehicle is used to establish the initial parking orbit.

These data illustrate a trend similar to that noted in Figure 6.14. Hydrogen operation must be at a specific impulse which is 200 seconds higher than that of ammonia in order to achieve comparable payload capabilities. Consequently, hydrogen performance will be superior to that of ammonia if compared on the basis of the maximum specific impulse attainable. At the .5 lbs. thrust level, hydrogen performance is 14% greater than that of ammonia.

6.4.4 Recommendations

It can be concluded, therefore, that hydrogen performance capabilities are superior to those of ammonia both at the level of performance specified by the current 30 KW engine development program and at the maximum specific impulse levels attainable in growth versions of the engine. It is recommended, therefore, that hydrogen be utilized as the propellant for the 30 KW arc jet propulsion system.

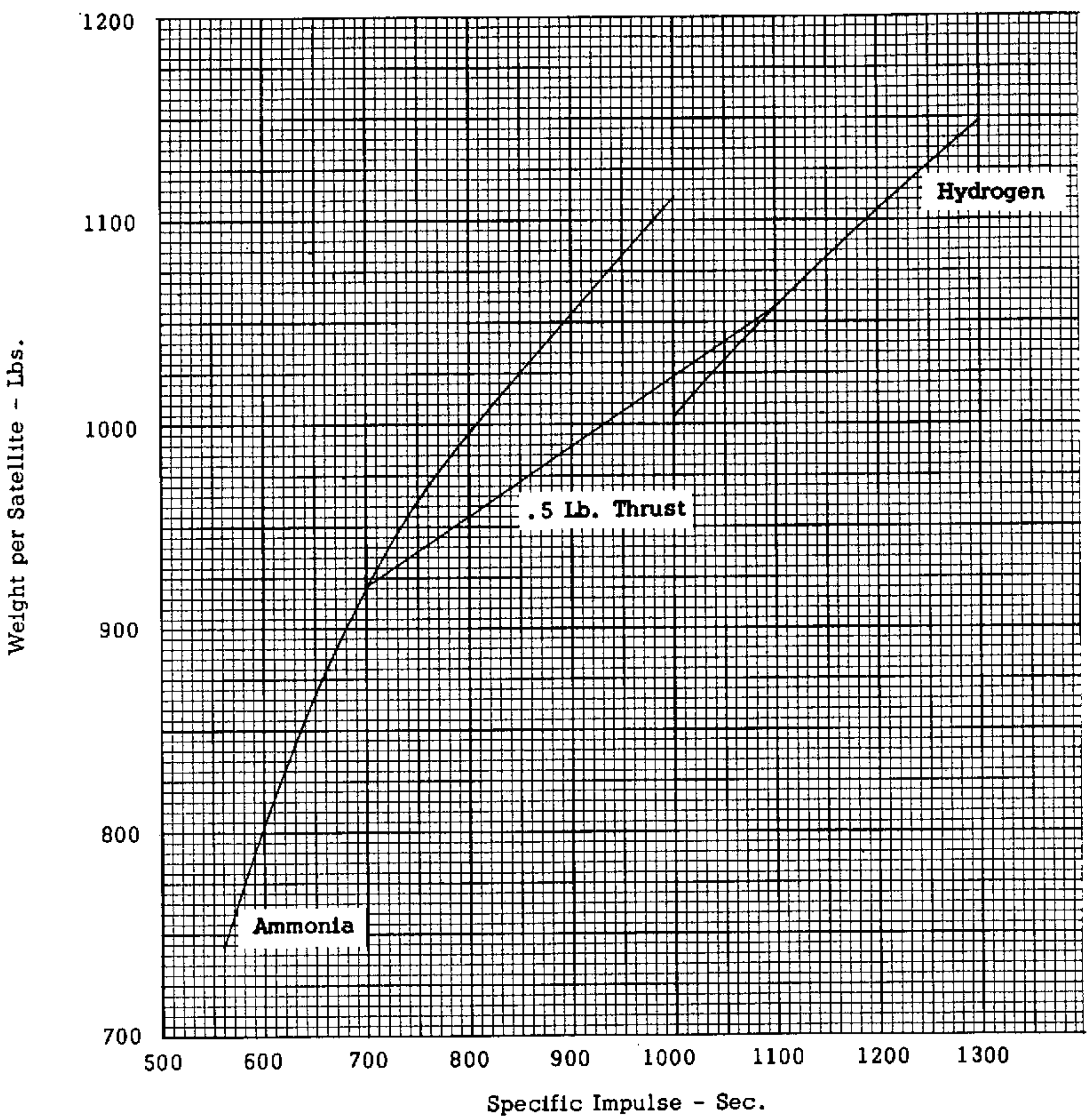

Figure 6.15 Satellite Distribution Mission Payload Capabilities. Saturn C-1 Booster to 300 M. Parking Orbit and 30 KW Arc Jet to 6000 M. Distribution Altitude. 10 Satellite Network.

7. SPACE PROGRAM APPLICATIONS

Current and projected NASA, DOD, and Commercial space program plans have been examined and evaluated in order to identify those programs most suitable for application of the Snap 8 - 30 KW Arc Jet Propulsion System. Particular emphasis has been placed upon investigation of those programs scheduled for operation in the 1965-67 time period which involve space vehicles of the Centaur or Saturn C-1 size. This time period is, of course, dictated by the March 1965 date currently programed for the initial flight of the Snap 8 Power System.

The programs described in the following sections, are believed to afford the most atttractive and logical applications for the subject propulsion system.

7.1 Advent

The most attractive application for the 30 KW Arc Jet Propulsion System is believed to be the Advent program. As currently constituted, Advent is a program for the development of an active military communication satellite. It involves the demonstration of the ability to perform a communication relay function from an equatorial, 22,240 mile altitude orbit. Successful demonstration of such capabilities should lead to the development and maintenance of an operational satellite system of three orbiting satellites.

The current Advent program utilizes the Atlas-Centaur boost vehicle to establish the satellite in the desired stationary orbit. The current payload capabilities of the Centaur - 1200 lbs. - appear to match the Advent requirements. There is, however, no margin for either growth or error.

The 30 KW Arc Jet could deliver the Advent payload to its required stationary orbit. A conception of an arc jet vehicle suitable for this purpose is illustrated in Figure 7.1. In this configuration, the Advent is stored within the cavity formed by the Snap 8 radiator in its stored position. The illustration shows the Advent being ejected through the side of the arc jet vehicle. Payload capabilities of 2000 lbs. are feasible with such an arrangement - thereby providing substantial Advent system growth capabilities.

An alternate arrangement is illustrated in Figure 7.2. In this configuration, the arc jet engine is ejected from the vehicle and the Snap 8

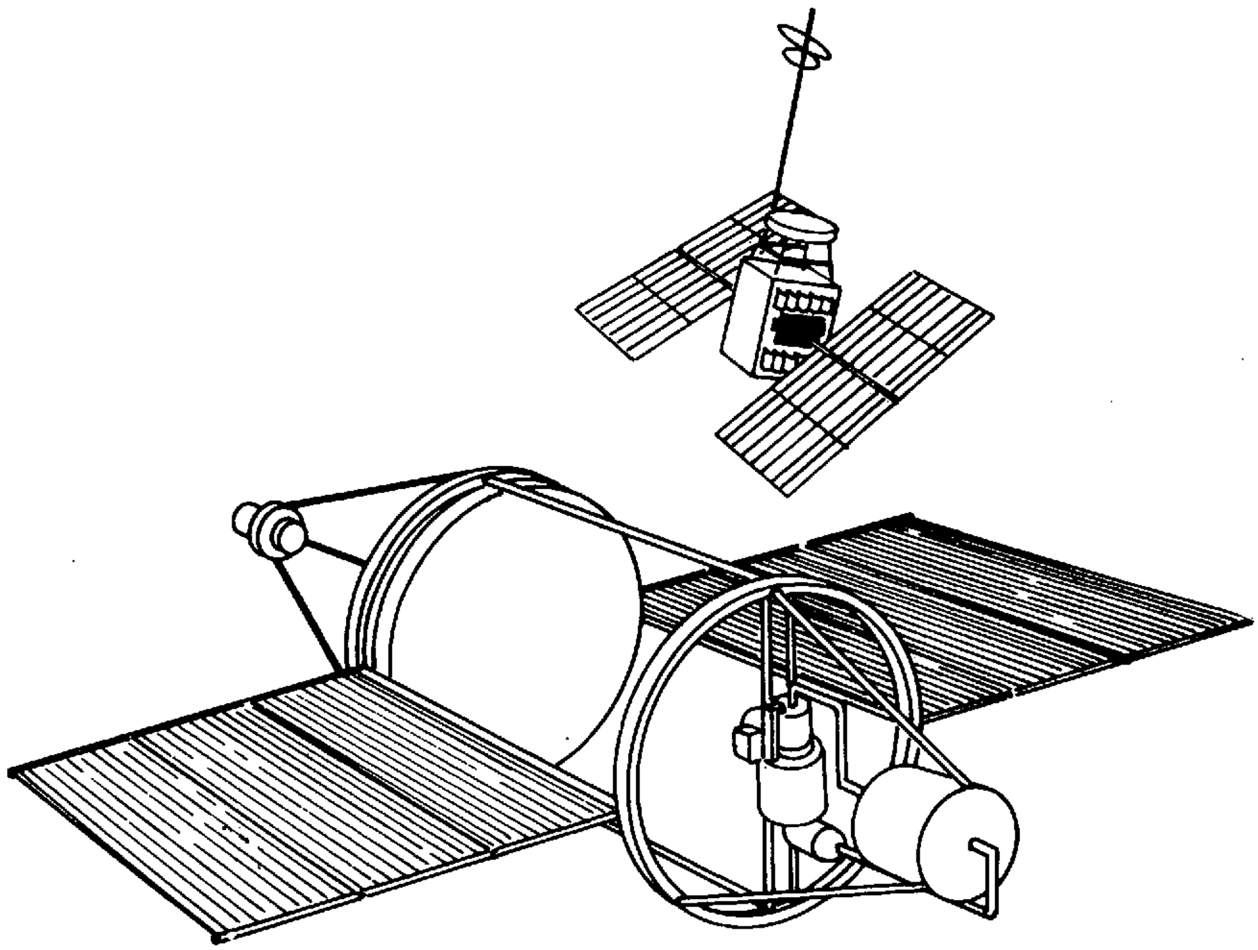

Figure 7.1 Advent Vehicle Injected into Orbit with 30 KW Arc Jet Propulsion System.

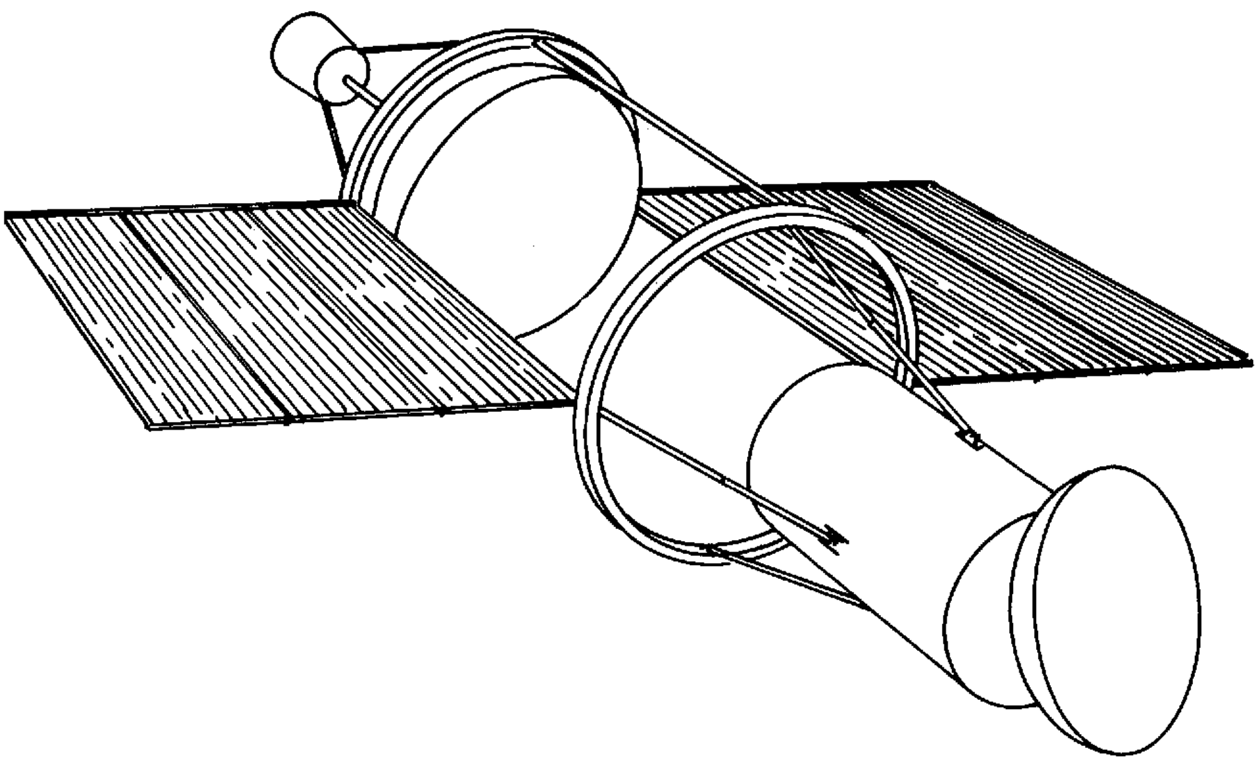

Figure 7.2   Advanced Advent Vehicle with 30 KW Power Capability.

power system retained to provide power for operating Advent. In the illustration, Advent is shown in an extended position behind the Snap 8 radiator. This configuration contains a power capability of 30 KW in the 2000 lb.. Snap 8 plus an additional weight capability of 2000 lbs. for communication equipment.

7.2 Relay II

Relay II is a NASA program for the development of a non-military active communication satellite. Orbit and payload requirements are expected to be quite similar to those of Advent. Although this program is not as clearly defined as Advent at the present time, the arc jet propulsion system appears to be equally suited for application to Relay II.

7.3 Commercial Comsat

There are, at the present, two basic approaches being proposed for a Commercial Comsat. One concept utilizes three satellites in a stationary satellite orbit and is, therefore, quite similar to the Advent and Relay II concepts. This concept has been proposed by Lockheed. Hughes is exploring a variation of this concept utilizing 24 hour non-equatorial orbits. Arc Jet system payload capabilities for this orbit configuration are 2800 lbs. plus the 30 KW Snap 8 power system.

The other concept involves the use of a large satellite network at the 4000 to 7000 mile altitude level. The G. E. approach uses a 10 satellite network and the A. T. &T. approach is considering a 30 to 50 satellite network. Individual satellite weights are in the 1000 to 1200 lb. size. Ten - 1200 pound satellites can be distributed at an orbit altitude of 7000 miles by the arc jet propulsion system from a 300 nautical mile parking orbit established by the Saturn C-1 boost vehicle. Thus, substantial reductions in the number of launches required can be obtained by the arc jet propulsion system.

There is, at the present time, no clear indication of which of these various concepts will be developed to an operational basis. The arc jet propulsion system, however, appears to be capable of a substantial contribution to each of the above programs.

7.4 Military Satellite Networks

Midas and Samos are currently being developed for application to an early warning system and to a photo reconnaisance system respectively.

Both systems utilize a 300 nautical mile orbit. Potential applications should materialize for the arc jet propulsion system if either of these programs are developed to the state of an operational satellite system of 3 or more satellites. Using the current vehicle requirements of 3000 lbs. for Midas and 4100 lbs. for Samos, the arc jet propulsion system could distribute 5 Midas vehicles or 3 Samos vehicles with a single Saturn C-1 launch.

7.5 Aeros and Astrostat

Aeros is an advanced version of the Nimbus weather satellite which involves a three satellite network in a 24 hour non-equatorial orbit. Astrostat is an advanced version of the OAO (Orbiting Astronomical Observatory) which involves a single satellite in a 24 hour equatorial (stationary) orbit.

The 700 lb. payload requirements for Aeros can be provided by either the Centaur booster or the arc jet propulsion system. There would, therefore, appear to be no incentive to develop the arc jet propulsion system for this application. If, however, it were under development for Advent or Relay II there might be some incentive for application to this program as well.

Astrostat, on the other hand, appears to require precision orbit and attitude stabilization that is beyond the current state of the art. The additional payload capabilities and the availability of low thrust for extended periods of time after the station orbit has been achieved may provide the additional capabilities needed for meeting these specifications. Here again, the acceptance of the arc jet propulsion system for Astrostat appears to be contingent upon prior acceptance as a part of the Advent or Relay II programs.

7.6 Surveyor

The only unmanned lunar exploration program currently planned for the 1965 to 1967 time period is the Surveyor. The original Surveyor concept called for a soft lunar landing of several hundred pounds of equipment. The payload would be utilized to bore into the lunar surface in order to obtain a sample and to analyze the sample and telemeter the results back to earth.

Current indications are that a growth version of the Surveyor will be developed to return a small sample of the lunar surface to the earth. This task had originally been scheduled for the Prospector program which appears to have been cancelled.

Arc jet capabilities for this dust return version of the Surveyor have been studied in some detail because of the importance of this program to the Apollo program. Soft lunar landing payload capabilities for several booster configurations are indicated in Table 7.1. The capabilities of a 30 KW shared-tank arc jet vehicle configuration are somewhat better than a three-stage Atlas - Centaur but not as good as a four stage Atlas - Centaur vehicle.

A 30 KW arc jet vehicle used with the Saturn C-1 booster can provide better payload capabilities than either a three or four stage Saturn C-1 vehicle. It is doubtful, however, that such large payload capabilities will be needed for the dust-return Surveyor.

The 30 KW arc jet would, therefore appear to have nothing to offer for the Surveyor program. This conclusion is, of course, subject to change if the performance capabilities of Centaur slip or if the payload requirements increase markedly.

8.0 References

1. D. G. Driscoll, "Cryogenic Tankage for Space Flight Applications", Linde Company, Presented at the 1959 Cryogenic Engineering Conference.

2. R. L. Bjork, "Meteoroids Versus Space Vehicles", ARS Preprint 1200-60, 1960.

3. W. R. Lloyd, "Radiator Design for Space Power Supplies", R60FPD224, Flight Propulsion Laboratory Department, Cincinnati, 1960.

4. "XT-761 30 KW Plasma-Jet Rocket Engine", Second Quarterly Progress Report, Flight Propulsion Laboratory Department, Cincinnati, April, 1961.

5. "XT-761 30 KW Plasma-Jet Rocket Engine", Third Quarterly Progress Report, Flight Propulsion Laboratory Department, Cincinnati, July, 1961.

6. H. Brown, "Computer Program for Space Trajectory Analysis of Electrical Propulsion Systems", R61FPD278, Flight Propulsion Laboratory Department, Cincinnati, 1961.

7. H. Brown and J. R. Nelson, "Thrust Orientation Patterns for Orbit Adjustment of Low Thrust Vehicles", ARS Journal, Vol. 30, No. 7, July 1960, pp. 635-637.

8. T. N. Edelbaum, "Propulsion Requirements for Controllable Satellites", ARS Journal, Vol. 31, No. 8, August 1961, pp. 1079-1089.

9. R. M. L. Baker; G. B. Westrom, C. G. Hilton; J. L. Arsenault; R. H. Gersten and E. Browne, "Efficient Precision Orbit Computation Techniques", ARS Preprint 856-59, June 1959.

9. Appendix

The low vehicle thrust-weight ratios associated with electrical propulsion and the attendant long propulsion periods required to complete most missions of interest severely complicate the process of trajectory and parametric mission analysis. The analytical solutions to the equations of motion which are needed in order to investigate the effects of propulsion and mission parameters can be obtained only by considerable simplification of the basic equations. The results obtained in this fashion are of unknown accuracy.

Specific solutions to these equations can be obtained by numerical integration of the basic equations of motion with a high speed digital computer program. The most effective numerical integration procedure involves the use of some form of the Variation of Parameters method (9) in order to maximize the propulsion time interval size per integration, thereby minimizing computer costs per trajectory. The minimum computer costs for trajectory calculations will still be too high to permit calculations of all trajectories required for a comprehensive parametric mission analysis.

A combined analytical and numberical approach was, therefore, developed. Precision trajectories were calculated for each of the candidate missions for two to three selected combinations of propulsion and mission parameters. At the same time, the basic equations of motion were simplified and solved analytically. Correlation factors were retained wherever simplifications were performed. The numerical results obtained from the computer calculations were then substituted into the resulting analytical expressions and utilized to evaluate the correlation factors. This process was repeated with differenct simplified forms of the basic equations until correlation factors were obtained for each of the candidate missions which were insensitive to variations in the propulsion and mission parameters.

The resulting empirical equations were then utilized, along with the correlation factors obtained, to generate the parametric mission performance curves presented in section 6. These equations can be utilized to generate additional parametric data if required. The equations obtained for each of the candidate missions are included in the following sections.

9.1 Satellite Transfer

The equations developed for the satellite transfer mission provide an accurate, reliable, and efficient means of determining propulsion time requirements for both altitude transfer operations and combined inclination correction-altitude transfer. The propulsion times obtained can then be utilized to determine overall payload capabilities.

The thrust orientation angle, with respect to the local horizontal, required for a combined inclination correction and altitude transfer is:

$$\tan\alpha = \frac{\pi \Delta i}{2\eta_4 \ln\sqrt{P_o/r_a}} \tag{9.1}$$

Propulsion time requirements for the initial propulsion period are:

$$t_1 = \left[\frac{I_{sp}W_o}{T}\right]\left[1 - \exp.^{A}\right] \quad \text{where } A = \frac{\sqrt{p_o/p_1}\ -1}{\eta_1 K_o g_o I_{sp}\cos\alpha} \tag{9.2}$$

The elements of the transfer orbit developed during this propulsion period are:

$$p_1 = r_a\left[1 - \frac{2\eta_2 g_o r_a^2 T\cos\alpha}{fM_e W_o(1-Tt_1/W_o I_{sp})}\right] \tag{9.3}$$

$$e_1 = 1 - \frac{p_1}{r_a} \tag{9.4}$$

Note that equations (9.2) and (9.3) must be solved interatively. A good first approximation for t_1 can be obtained by using $r_a = p_1$.

The final propulsion period required for converting the transfer orbit into the desired circular orbit can be obtained from:

$$t_2 = \frac{W_1 e_1(1 + .25e_1)}{2\eta_3 K_2 g_o T(1-Tt_2/2W_1 I_{sp})} \tag{9.5}$$

This second propulsion period will, in general, be quite small in comparison with the initial propulsion period and may be neglected.

Since the total impulse - product of thrust and propulsion time - is dependent primarily on the engine specific impulse, calculations may be performed for a single thrust level and the results used to generate the requirements for other thrust levels of interest.

Evaluation of the preceding equations indicate that they are approximately equivalent to:

$$\Delta V = \frac{\sqrt{fM_e}}{\cos\alpha}\left[\frac{1}{\sqrt{p_o}} - \frac{1}{\sqrt{r_a}}\right] \tag{9.6}$$

where the velocity increment - ΔV - can be utilized with the conventional rocket equation:

$$\Delta V = g_o I_{sp} \ln (W_o/W_2) \tag{9.7}$$

to determine propellant requirements and, therefore, propulsion time. For an altitude transfer with no inclination correction, equation (9.6) reduces to the difference between the initial and final circular orbit velocities. The results obtained from equation (9.6) appear to be within 1.5% of those obtained using equations (9.1) through (9.5).

9.2 Lunar Transfer

The equations developed for the lunar transfer are somewhat more cumbersome than the preceding set for the satellite transfer because of the necessity for transforming the coordinate system from an earth reference to a moon reference. The equations indicated consider the ellipticity of the moon's orbit about the earth and the earth-moon perturbations during the coasting phase of the trajectory. Inclination corrections have been assumed to be negligible.

The requirements for the initial propulsion period can be obtained from:

$$t_1 = \left[\frac{I_{sp}W_o}{T}\right]\left[1 - \exp.^{C}\right] \quad \text{where } C = \frac{\sqrt{p_o/p_1} - 1}{\eta_6 K_o g_o I_{sp}} \tag{9.8}$$

$$e_1 = \frac{\eta_7 g_o T p_1^{\,2}}{fM_e W_o(1 - Tt_1/W_o I_{sp})}\left[\frac{2 + e_1\cos\phi_1}{(1 + e_1\cos\phi_1)^3}\right] \tag{9.9}$$

$$p_1 = (1 - e_1) r_3 \tag{9.10}$$

An iterative solution is required here, as well. A true anomaly at the end of the propulsion period - ϕ_1 - of 72° should be utilized.

The effect of earth-moon perturbations have been correlated as:

$$a_1 = \frac{p_1}{1-e_1^2} \tag{9.11}$$

$$a_2 = a_1\sqrt{\frac{r_3}{p_1}} \tag{9.12}$$

$$p_2 = p_1\left[\frac{194{,}000}{a_1}\right] \tag{9.13}$$

The above equations (9.12) and (9.13) are based upon empirical observations of results obtained with minimum energy transfer trajectories and the 30 KW arc engine propulsion characteristics. They should be used with caution for propulsion and trajectory parameters outside the range of the above conditions.

The transfer orbit elements can be transformed to a moon reference system by:

$$V_1 = \sqrt{fM_e\left[\frac{2}{r_1} - \frac{1}{a_2}\right]} \tag{9.14}$$

$$\sin\mu_1 = \frac{\sqrt{fM_e p_2}}{r_1 V_1} \tag{9.15}$$

$$\cos\phi_3 = \frac{r_1^2 + r_3^2 - r_2^2}{2r_1 r_3} \tag{9.16}$$

$$V_2 = \sqrt{V_1^2 + V_3^2 - 2V_1V_3 \sin(\mu_1 - \phi_3)} \tag{9.17}$$

$$a_3 = \frac{fM_m r_2}{2fM_m - r_2V_2^2} \tag{9.18}$$

Propulsion requirements appear to be minimized by the use of a moon - earth - vehicle angle - ϕ_3 - of 0^o.

The propulsion requirements for the final phase of the mission can be obtained from:

$$\Delta V_f = \sqrt{\frac{fM_m}{r_4}} \pm \sqrt{\frac{fM_m}{a_3}} \qquad (9.19)$$

$$t_2 = \left[\frac{I_{sp}W_1}{T}\right]\left[1 - \exp. \; D\right] \quad \text{where } D = \frac{-\Delta V_f}{\eta_8 g_o I_{sp}} \qquad (9.20)$$

The positive sign of equation (9.19) should be utilized for negative values of a_3 and the negative sign for positive values of a_3.

The above lunar equations will result in a characteristic mission velocity of about 28,000 f.p.s for propulsion characteristics compatible with the arc jet engine. If the earth-moon perturbations are ignored, the mission requirements will appear to be about 27,000 f.p.s. It can be concluded, therefore, that the earth-moon interactions increase the characteristic velocity requirements by 1000. Calculations using two-body techniques would, therefore, be in error by about 2.5% and should be utilized only for preliminary feasibility investigations.

9.3 Satellite Distribution

Satellite distribution characteristics differ from those of the proceeding missions in that the total impulse requirement decreases with decreased thrust-weight ratio. It is this very characteristic that leads to the extremely attractive payload capabilities of low-thrust electrical propulsion systems in comparison with comparable chemical rocket capabilities.

Propulsion requirements for each satellite distribution operation can be obtained from the equation:

$$t = \sqrt{\frac{4W_o p_o \Delta l}{3\eta_5 g_o T}} \quad \text{where } \Delta l = \frac{2\pi}{n} \qquad (9.21)$$

The initial vehicle weight for each succeeding distribution operation can be determined by subtracting the weight of one satellite and the propellant weight associated with the results of equation (9.21). Individual satellite weight must be iterated until the final weight after distribution of n satellites equals the desired equipment weight.

For satellite distributions in excess of 10 satellites, the above procedure becomes extremely tedious. Satellite distribution capabilities for such large networks can be obtained from the equations:

$$W_p = \frac{1}{I_{sp}} \sqrt{\frac{8 \pi p_o W_o T}{3 g_o n}} \tag{9.22}$$

$$W_s = \frac{W_o - W_e - 1.2 W_p \left[(n-1) - n(n-1)(n-2) W_p / 4W_o \right]}{1.06n - n(n-1)(n-2) W_p / 4W_o} \tag{9.23}$$

9.4 Nomenclature

- a_1 - semi-major axis at end of transfer propulsion - miles
- a_2 - semi-major axis before transition to moon's field - miles'
- a_3 - semi-major axis at beginning of lunar capture - miles
- e_1 - eccentricity of coasting orbit
- f - gravitational constant - $9.404(10)^{-14} m^3/lb.hr.^2$
- g_o - gravitational constant - 79,020 $m/hr.^2$
- I_{sp} - specific impulse - hrs.
- K_o - $\sqrt{p_o/M_e}$
- K_2 - $\sqrt{r_a/fM_e}$
- M_e - earth mass - $1.3177(10)^{25}$ lbs.
- M_m - moon mass - $1.6204(10)^{23}$ lbs.
- n - Number of equally spaced satellites to be distributed
- p_o - initial semi-latus rectum-miles

p_1 - semi-latus rectum of coasting orbit-miles

p_2 - semi-latus rectrum before transition - miles

r_a - desired final orbit radius-miles,

r_1 - vehicle-earth distance-miles

r_2 - vehicle-moon distance - 23,000 miles

r_3 - earth-moon distance - 238,138 miles

r_4 - final orbit radius about moon-miles

t - propulsion period for each satellite - hrs.

t_1 - initial propulsion period - hrs.

t_2 - final propulsion period - hrs.

T - thrust-lbs.

V_1 - vehicle-earth relative velocity-mph.

V_2 - vehicle-moon relative velocity-mph.

V_3 - moon-earth relative velocity - 2270 .8 mph.

W_o - initial vehicle weight - lbs.

W_1 - vehicle weight during coasting - lbs.

W_e - power supply plus engine weight - lbs.

W_p - propellant weight - lbs.

W_s - individual satellite weight - lbs.

α - thrust orientation angle to orbit plane

Δi - inclination correction - radians

Δl. - longitude correction per satellite

ΔV_f - effective velocity increment of lunar capture - mph.

ΔV - mission characteristic velocity - mph.

η_1 - correlation factor - 1.0043

η_2 - correlation factor - 1.0000

η_3 - correlation factor - 1.0000

η_4 - correlation factor - 1.0037

η_5 - correlation factor - 1.0013

η_6 - correlation factor - .9956

η_7 - correlation factor - .8541

η_8 - correlation factor - .9800

μ_1 - angle between earth-vehicle line and vehicle-earth relative velocity

ϕ_1 - true anomaly of earth relative orbit - 72°

ϕ_3 - angle between earth-moon and earth-vehicle line